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THRUST CHAMBER COOLING TECHNIQUES FOR SPACECRAFT ENGINES

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FINAL REPORT

THRUST CHAMBER COOLING TECHNIQUES FOR SPACECRAFT ENGINES

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VOLUME I

EVALUATION PROCEDURE AND ANALYSES

Contract NAS 7-103

Project 278

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I. SUMMARY

Space missions envisioned for liquid propellant rocket engines encompass a wide spectrum of performance and structural requirements. Thrust levels from a few pounds to many thousands of pounds per engine and run times from fractions of a second to many minutes may be required. Installations vary from those in which the engine is free to radiate heat to space to those in which the engine must be buried within the vehicle. The most promising propellants include the storable hypergolics as well as the cryogenic high energy combinations.

All of these spacecraft engines have one problem in common: The energy generated by the propellants must be contained and the surrounding structure must be protected. The materials involved must be able to withstand the high temperature of the combustion gases or must be cooled to safe operating temperatures.

Thrust chamber cooling concepts developed to cope with these requirements either singly or in combination include regenerative or convective cooling, radiation cooling, film or transpiration cooling, ablation, and inert or endothermic heat sinks.

This report is composed of two volumes and it presents a study of the range and limits of applicability of each of these cooling concepts and procedures for selecting and designing the most suitable cooling system for a specific spacecraft engine application.

Volume I of this report outlines the procedure proposed for evaluating the cooling requirements for a liquid rocket space engine and provides analyses and data for selecting the applicable and best cooling techniques. Four specific examples of propulsion requirements are used to demonstrate the cooling technique selection procedure.

Volume II of this report presents thrust chamber design procedures for each cooling technique, including design data for propellants and thrust chamber materials, as well as additional details of mission requirements and a bibliography arranged by subject entries.

It is the hope of the authors that this report will be useful for several years to come. It is realized, however, that the work presented here is subject to constant updating as a function of new materials and fabrication capabilities and design techniques. The bibliography presented in Volume II should be used to supplement this report by providing additional detail design and test data for specific areas of interest.

Technical areas requiring continued intensive research and development include high temperature refractory material systems for uncooled nozzle throat inserts and the application of film and transpiration cooling to high pressure, high temperature, corrosive combustion gas propulsion systems.

II. INTRODUCTION

A. Program Objectives

A program to facilitate the selection and design of the most suitable cooling method for various spacecraft liquid rocket engines has been sponsored by the National Aeronautics and Space Administration, Office of Liquid Rockets, under Contract NAS 7-103.

The objectives of the program conducted under this contract were:

1. To determine the applicability and limitations of the various thrust chamber cooling methods for liquid propellant rocket engines used to fulfill spacecraft propulsion requirements

2. To present thrust chamber design procedures for each cooling technique and to provide a basis for comparing different cooling designs on the basis of applicability, weight, and performance

3. To develop and present a rapid and convenient procedure for selecting the most suitable cooling method for the various spacecraft engine applications

B. Scope of Mission Requirements and Engine Types

The scope of space missions and engine types considered includes missions that can be carried out with Centaur, Saturn, and Nova class vehicles. The engine applications include those which would provide the propulsion needed to accomplish orbital or trajectory correction, orbital rendezvous, and lunar and planetary landing and takeoff. The engine types have been limited to those using liquid propellants. Engine sizes considered in detail have been those in the 100 to 20,000 pound thrust class, although the results and conclusions apply over a much wider range of sizes.

C. Program Approach

The technical approach employed to accomplish the objectives of this program has been to evaluate each available cooling technique to define its range of application and the nature of the limitations of its applicability. The currently available experimental data and technical data on mission requirements, propellant performance, cooling systems, and structural materials have been evaluated for their relationship to the selection of a cooling technique and design of a rocket engine thrust chamber.

Parameter studies have been conducted to define the range of capabilities of each cooling method and to permit a comparison of different cooling methods for a particular application.

D. Effect of State of the Art Advances on the Results of the Program

An attempt has been made in presenting the results of this program to provide for advances in the state of the art in the next several years in the areas of improved materials, propellants, or new mission requirements. This has been done by including the basic thrust chamber design procedures in detail, and by including in the parameter studies a range of variables beyond the current material capabilities.

It seems probable that the new advances in cooling techniques will come about by an optimization of combined cooling techniques. These advances may well be in the area of combining a form of film cooling with one of the other cooling techniques.

E. Cooling Techniques Studied

The cooling techniques evaluated during this program have included the following:

1. Regenerative cooling
2. Radiative cooling
3. Ablative cooling
4. Film cooling
5. Transpiration cooling
6. Inert heat sink
7. Endothermic heat sink
8. Open tube convective cooling (dump cooling)
9. Combinations of the above

F. Sources of Data

Data and analyses relating to these cooling techniques have been gathered from the large amount of work done in these areas at Marquardt as well as by other agencies, both government and private. Much of this has already been published in unclassified literature. Various material vendors have been most generous in supplying material data as well as test results. Some of the published experimental data found useful in the evaluation of cooling techniques are still classified. References to the more useful classified data are presented in the bibliography included in Volume II.

G. Specific Design Studies

In order to check out the design and selection procedures presented in this report, four specific thrust chamber design studies were completed and they are presented in Section VIII of this volume. These studies include the following examples:

1. A variable thrust, earth storable propellant, deep space engine

2. A constant thrust, oxygen-hydrogen fueled space engine

3. A constant total impulse engine with firing time and thrust as parameters

4. A constant thrust, space storable propellant, deep space engine

H. Limitations and Interpretations of Results

Even as this report is written, several agencies, including Marquardt, are developing and evaluating new materials and several novel cooling concepts. The optimization and determination of the ultimate limits of these new techniques will take several years. Therefore, any limitations and optimizations presented in this report are subject always to change due to these advances in the state of the art. Therefore, this report defines the nature of these limitations, such as limitations due to material properties or a certain assumed component geometry. If these can be improved, obviously the same limits would not apply. The results of these studies should be interpreted accordingly.

III. SUMMARY OF PROCEDURE FOR SELECTION OF A THRUST CHAMBER COOLING METHOD

The thrust chamber cooling method selection procedure presented in this report is intended to facilitate the design or specification of the most suitable thrust chamber cooling method to fulfill a given space propulsion requirement. This procedure will establish which cooling techniques are applicable to various portions or components of a liquid rocket thrust chamber. Of the applicable techniques, an optimum choice may then be made on the basis of weight, performance penalty, or other factors such as cost, margin of safety, development costs, etc.

Steps presented for the selection of a cooling method are as follows:

1. Specification of propulsion requirements (Section IV)
2. Screening and review of various cooling techniques for applicability (Section V)
3. Completion of a preliminary thrust chamber weight analysis for applicable cooling methods (Section VI)
4. Evaluation of propulsion performance penalties (Section VII)
5. Selection of one or more promising cooling techniques for a more complete design study (Section VIII)

The initial selection procedure outlined in this section can be carried out with a minimum of analysis and calculation. Optimization and final choice between two or more applicable thrust chamber designs may be based finally on factors beyond the scope of this report. Detailed design considerations and cooling limitations are covered in Section III of Volume II.

IV. PROPULSION SYSTEM SPECIFICATION

The initial specification of the propulsion system may be quite general as derived from a space mission analysis. Thus, the initial requirement may be given as an initial spacecraft mass and a velocity change or as a thrust and burning time. This, of course, leaves many questions to be answered before a thrust chamber design can be chosen. If these are the only data given, then several designs would have to be carried out far enough to establish the advantage of one propulsion system over another.

For the purposes of this report, as many as possible of the propulsion system requirements outlined below should be specified uniquely or in terms of limits.

A. Mission Requirements (Engine Purpose)

Specifying the purpose of the engine establishes several important cooling parameters such as the engine location, thrust level, burn time, and duty cycle. A large number of spacecraft missions and propulsion requirements are summarized in Table I. Typical engine requirements from Reference 1 for thrust variability, restart, service life, duty cycle, thrust level, and engine location are presented. Of particular interest are the run times which range, in general, from 40 to 400 seconds with many maneuvers requiring burn times of less than 100 seconds. This is shown graphically in Figure 1 which presents thrust and run time versus total impulse requirements from the data of Table I. It may also be desirable for the same engine to fulfill more than one type of mission or to be reused on subsequent missions.

Thrust versus time relationships for different types of maneuvers are shown in Figure 2. Typically, the thrust-time requirements are different for lunar landing, orbital rendezvous, attitude control and lunar takeoff as shown. Section IV of Volume II presents further detailed considerations of the space mission maneuvers included in Table I and how they affect propulsion and cooling requirements.

Thus, definition of the engine purpose is the first step in establishing the requirements for chamber design and cooling techniques.

B. Propellants

Specification of the propellants is the next step in establishing the thrust chamber design requirements. The choice of propellants may be based on a specific impulse requirement to accomplish a given space mission. Also, the choice will be strongly affected by the current state of the art with respect to combustion experience, handling, availability, etc. With regard to cooling method, the propellant choice determines the combustion gas temperature and the gas composition. Several propellants are excellent as coolants while others have little cooling capability. The high temperature combustion gas constituents vary widely in their compatability with candidate thrust chamber materials. These factors are evaluated in detail later in this report. The liquid propellants considered in this report as typical of the basic classes of propellants include the following:

Oxidizers	Fuels
N_2O_4	N_2H_4
O_2	H_2
OF_2	0.5 N_2H_4-0.5 UDMH
F_2	B_2H_6 and CH_4

These propellants, their performance, and their properties are covered in Section IV of Volume II of this report.

C. Propulsion Requirements

After specifying the engine purpose or mission and the propellant combination, the remaining engine propulsion requirements should be detailed as completely as possible in terms of the following items:

1. Total impulse
2. Velocity change
3. Thrust level (as a function of time, if possible)
4. Run time
5. Throttling range
6. Number of restarts
7. Impulse cut-off accuracy
8. Pulse repetition rate
9. Minimum impulse bit
10. Number of engines
11. Minimum I_{sp}
12. Vectoring requirement
13. Thrust chamber pressure limits (or propellant supply pressure)

D. Environmental and Operational Requirements

As many as possible of the following engine and spacecraft characteristics should also be specified with respect to their effect on the thrust chamber design:

1. Engine location with respect to the spacecraft structure
2. Engine envelope limitations
3. Engine configuration (C-D or plug nozzle, contraction ratio, L*, expansion ratio, etc.)
4. Exterior temperature limits or heat loss limits
5. Oxidizer/fuel ratio
6. Storage time in space
7. Distance and attitude of spacecraft with respect to the Sun
8. Maximum acceleration and vibration loads
9. On-board nuclear emission
10. Re-entry environment
11. Reliability requirements
12. Ground check-out requirements

V. GENERAL APPLICABILITY CHARACTERISTICS OF THRUST CHAMBER COOLING METHODS

Certain of the propulsion system requirements specified in the foregoing section directly affect cooling and may strongly favor one or more cooling methods while wholly eliminating others. Also, the severity of the cooling requirement will vary over a wide range from inside the combustion chamber, through the exit nozzle throat, and along the exit cone or skirt. Hence, the optimum thrust chamber design may well incorporate two or more basic cooling methods, either combined or applied separately to the different chamber components.

A preliminary screening to determine applicable cooling techniques may be accomplished by consideration first of some of the more critical propulsion requirements and their effect on cooling techniques as pointed out below. A screening chart summarizing these general design factors is presented in Figure 3. The screening chart shows, for each cooling method, whether or not an operating requirement or range of application may be a limiting factor. A more detailed discussion of these factors is presented in the text of this section, first in terms of the propulsion requirement, then in terms of the limitations on each cooling method.

From these initial screening steps, one or several thrust chamber design approaches may appear promising. A preliminary layout of these designs along the lines shown in Figure 4 will permit a weight study to be made as outlined in Section VI.

A. Cooling Techniques Applicable to Particular Propulsion Requirements

1. Propellant Selection

Cooling techniques applicable to the different classes of propellants such as the earth storable hypergolics, the cryogenics with hydrogen as fuel, and the space storable combinations with the OF_2 as oxidizer, are presented in Table II. The relative severity of the cooling problem is indicated in the table by the flame temperature, the principle exhaust products, and the regenerative cooling capability of the propellants.

The applicability envelope for regenerative cooling of four propellant combinations is presented in Figure 5 as a function of chamber pressure and thrust level.

For the earth storable propellants in the chamber pressure range below 250 psi, the choice of cooling techniques applicable, includes regenerative, radiative and ablative cooling. Also, for short run times, the use of a heat sink design is possible. For higher pressures and long run times, film or transpiration cooling may be required.

For the cryogenic propellants using liquid hydrogen as the fuel, convective cooling is attractive because of the excellent heat transfer properties of hydrogen. Hydrogen may be used as a regenerative coolant and also as a film or transpiration coolant. On larger engines (10,000 pounds thrust and greater) dump cooling or open tube cooling requiring only a fraction of the hydrogen may be used effectively in nonregenerative convective cooling. Radiation, ablative, and heat sink cooling are also applicable so that some optimum combination of these cooling techniques will probably provide maximum engine performance and flexibility with minimum complexity.

For the space storable propellants using the OF_2 oxidizer, the high flame temperature and the oxygen containing exhaust products provide the severest of material environments. The flame temperature exceeds the melting temperatures of the most refractory of the metals and carbides. Radiation cooling would be applicable to the combustion chamber only at very low chamber pressures or in the exit nozzle skirt at large expansion ratios. None of the propellants in this group are suitable for convective cooling. Ablative materials would be suitable in the combustion chamber and exit skirt for limited run times. In the nozzle throat region, the heat sink concept using a material such as pyrolytic graphite or impregnated porous tungsten is the most suitable for limited run times. For longer run times, film and transpiration cooling would be applicable with a suitable coolant. The capabilities of these propellants for this application have not been evaluated. Some auxiliary inert coolant may be required for some applications.

2. Pulsing Requirement

If rapid on and off cycling of the engine is required, passive protective techniques are best. Starting and stopping of coolant flow is likely to limit response time or cause excessive coolant waste in a film cooled engine in addition to giving rise to residual thrust from excess coolant exhaust.

Applicable Cooling Techniques

Radiative

Heat sink (Inert)

Ablative (Some residual thrust)

3. Long Run Time

Long run time implies a high propellant to hardware weight ratio. Minimum performance penalty is important.

Applicable Cooling Techniques

Regenerative

Radiative

Ablative (Weight increases as $[\text{run time}]^{1/2}$)

Open tube (Some performance penalty)

4. Throttling

The cooling requirement for throttling operations varies with chamber pressure as thrust is varied.

Applicable Cooling Techniques

Radiative

Ablative (Char rate almost independent of thrust)

Regenerative (Range of throttling limited)

Open tube (Coolant can be separately controlled)

Heat sink (Time limited)

Film cooling (May incur increased I_{sp} losses)

Transpiration cooling (May incur increased I_{sp} losses)

5. Fast Response

Accurate impulse control requires fast response of cooling technique and absence of residual thrust.

Applicable Cooling Techniques

Radiative

Heat sink

Ablative (Some residual thrust)

6. Limited Engine Envelope

For required total impulse or velocity change, the engine size may be reduced by employing a lower thrust engine for a longer time, by using a limited expansion ratio, or by employing higher chamber pressures.

Applicable Cooling Techniques

Regenerative

Open tube

Film

Transpiration

Ablative (Throat may impose pressure or time limit)

B. Applicability of Specific Cooling Techniques

1. Regenerative Cooling

a. Cooling Limitations

Three specific factors have been utilized to describe limitations on regenerative cooling of rocket thrust chambers. These are a coolant supply pressure requirement, a minimum practical passage dimension, and a maximum coolant temperature rise. The coolant temperature limitation is expressed either as a maximum nozzle expansion ratio which can be cooled, or in the case of hydrogen cooling, as a percentage of a maximum allowable enthalpy rise.

Methods by which these limits are derived and correlated with thrust and chamber pressure are explained in Volume II. Boundaries of the feasibility map for regenerative cooling with the propellant combinations of N_2O_4/N_2H_4, O_2/H_2, F_2/H_2, and N_2O_4/Aerozine 50 are presented in Figure 5. Reasonable cooling solutions are possible within these envelopes.

Further increases in chamber pressure over those shown in Figure 5 may be accommodated by resorting to supplementary methods such as film cooling, ceramic coatings, etc.

Nozzle wall temperatures, while not specifically expressed in any of the limiting envelopes, are nevertheless inherent in them. For the class of liquid coolants transferring heat by nucleate boiling, the chamber wall operating temperature is a fixed function of the coolant pressure. In the convective cooling situation, using hydrogen, all points in the grid were computed for a 2000°R wall surface temperature. This represents a realistic level for currently developed rocket engine construction materials.

b. Operational Limitations

Several factors are apparent that, while not directly limiting or excluding regenerative cooling, should be considered in the process of selecting a cooling method. In general, conclusions about these parameters can be made only after making complex tradeoff studies between engine weight, volume, design simplicity, reliability, etc.

(1). Restart

The regenerative cooling concept imposes no limitations upon restart of rocket engines other than added complexity to sequencing.

(2). Pulse Operation (Response Time)

Starting and stopping operations exhibit poor response if there is no valve between the coolant passages and injectors. While regenerative cooling should be able to satisfy "ΔV engine" requirements, attitude control or "station keeping" would seem too exacting.

(3). Space Storage (Purging)

In general, the volume of a liquid cooling jacket should be gas purged after each operating cycle. Some of the reasons for this are as follows:

1. Slow draining of jacket by evaporation of liquid coolants
2. Possible sporadic ignition of hypergolic propellants
3. Possible freezing of coolant in a space environment and blocking flow passages

(4). Throttling

Specific problems of throttling regenerative cooled engines are discussed in Volume II. Graphs illustrating throttling capabilities and statements concerning design concepts are presented. In general, the throttling ratio is limited and imposes restrictions on the regenerative cooling envelope of applicability.

(5). Propellant Choice

Hydrogen is the best coolant, followed by N_2H_4 and Aerozine 50 in that order. Not much is known concerning the capabilities of diborane. Pentaborane, however, has only limited cooling potential.

(6). Zero g

A weightless state should cause no important effects in regenerative cooling.

(7). Meteoroids

It is difficult to estimate the effect of a penetration of the cooling jacket by a meteoroid. Regenerative cooled chambers have been known to operate, without catastrophic results, with as much as 10 percent of the coolant passages containing holes. External leaks, in the atmosphere, can be quite serious. Whether they would represent anything other than a performance loss in space remains to be determined.

(8). Exterior Wall Temperature

Exterior wall temperatures would approach coolant temperature; less than 400°F for storable liquid fuels and temperatures above 1000°F for hydrogen.

2. Open Tube (Dump Cooling)

a. Cooling Limitations

Dump cooling is an attempt to make use of the excellent heat transfer characteristics of high temperature gaseous hydrogen. The object is to cool the chamber walls convectively with a very small percentage of the total hydrogen flow thereby eliminating the coolant jacket pressure drop in the main propellant flow. Since the majority of fuel never passes through the cooling jacket and that which does, is dumped to space at the nozzle exit, the maximum pressure to which the fuel need be raised is the injection pressure. This reduction in fuel pressurization represents the major advantage of the dump cooling concept.

It is of course obvious that a chamber that cannot be cooled with the total fuel flow by regenerative methods, cannot be cooled by a fraction of the fuel by dump procedures. Therefore, dump cooling is limited to those areas wherein regenerative cooling is relatively easy. In these regions of high thrust or low chamber pressure, the hydrogen coolant capacity heat is limited due to the coolant temperature approaching the maximum structural temperature.

Primary among the penalties involved in the dump cooling design, is the increase in hydrogen required. Most investigators report dump cooled designs using around 2% of the total propellant flow rate. At the normal O_2/H_2 mixture ratio of 5:1, however, this represents a 12% increase in hydrogen. With the very low storage density (from 4.5 to 5.0 pcf) for hydrogen, this can represent a significant amount of tank volume for large thrust chambers of long duration. To help counteract this penalty, the dump flow may be expanded to produce useful thrust at a level of I_{sp} slightly greater than that of the thrust chamber. The net performance effect is small and a system analysis would be required for complete evaluation.

In summation, there appears to be at least two potential uses for dump cooling of large thrust engines. The first is where the saving of fuel pressurization overcomes the increased tankage volume. The second is for short duration, pulse operation at pressures in excess of radiation cooling limits, where soak back and duty cycle considerations in ablative chambers would result in chamber weights greater than those for dump cooling. The weights of dump cooled chambers are taken to be the same as those for regeneratively cooled chambers.

b. Operational Limitations

(1). Restart

There are no limitations in restart operations with open tube cooling.

(2). Response Time

Open tube cooling has much faster response than regenerative cooling due to the reduced mass of the coolant.

(3). Space Storage (Purging)

The necessity for purging seems unlikely with open tube cooling due to the low mass of coolant, simple flow path, and the independent nature of the coolant jacket and injector.

(4). Throttling

Since coolant flow can be regulated independently, open tube cooling seems ideal for throttling.

(5). Propellant Choice

Open tube cooling is limited to gaseous coolants that are stable at high temperatures, i.e., hydrogen.

(6). Meteoroids

Penetrations in the expansion nozzle could result in askew thrust vectors. Otherwise, the situation would be similar to that for regenerative cooling with considerably less performance penalty.

(7). Thrust Levels

Open tube cooling generally is applicable only to large thrust engines (> 10,000 lbf).

3. Radiation Cooling

a. Cooling Limitations

The characteristic limitation on radiation cooling is the availability of materials which can operate at the equilibrium thrust chamber wall temperatures reached during steady state operation. These temperatures are most sensitive to chamber pressure and nozzle area ratio. Typical predicted equilibrium wall temperature distributions as a function of chamber pressure and nozzle area are shown for one propellant combination in Figure 6. Of particular interest, is the application of radiation cooling to the expansion nozzle skirt at large area ratios. Due to the reduced heat fluxes, low static gas pressures, and large surface areas involved, radiation cooling can be employed to gain increased engine thrust at small increases in structural weight. Radiation cooled chambers of thrust levels less than 100 pounds have been developed to run under steady state conditions for over an hour at chamber pressures of 90 psia. Experimental heat transfer rates in small thrust chambers can be controlled by injector design to permit chamber pressures well above theoretical limits.

The most important limit on firing duration is the life of the protective coatings used on refractory metals. Actual thrust chamber lives of several hours have been demonstrated with molybdenum disilicide at metal temperatures above 3000°F. The silicide coatings of other refractory metals are probably comparable, based on test samples in oxyacetylene and plasma flames. Data on time-temperature capabilities of coated refractory metals are presented in Figure 152 of Volume II. Very thin wall chambers might also have a duration limit due to creep.

Figure 7 presents a typical plot of limiting chamber pressure versus engine thrust based on a limiting throat wall temperature of 3300°F as calculated from normal heat transfer methods (Reference 2). The experimental point indicates the operating pressure of a 100 pound thrust radiation cooled molybdenum chamber with an L* of less than 15 inches. The typical throat wall temperatures for this thrust chamber are less than 3000°F.

The propellants establish very important limits of applicability, which depend on the compatibility of the combustion gas with the motor walls or coatings and the combustion gas temperature. Most of the propellant combinations considered contain water vapor as the most reactive gas, but F_2/H_2 and OF_2/B_2H_6 products are primarily HF, H_2, or other unusual species, many of which have not been completely evaluated as to their reactions with bare refractory metals and graphite. Since HF is not highly reactive with tungsten nor with graphite, a radiation cooled motor of bare tungsten or pyrolytic graphite is probably feasible for F_2/H_2 at some chamber pressures and mixture ratios. Thrust chamber materials and coatings for use with OF_2/B_2H_6 are not known at present.

b. Operational Limitations

(1). Space Vacuum

One hazard to operation in space is the possible evaporation of the protective coating when the hot motor is exposed to vacuum. This has not been found to be a serious problem for molybdenum disilicide, but the behavior of other coatings in a vacuum is not known.

(2). Earth Re-Entry

A radiation cooled thrust chamber can be operated during earth re-entry if it is situated so that its walls do not exceed the maximum coating temperature. A buried installation is also possible, using a cooled or heat sink radiation shield between the motor and the vehicle.

(3). Clustered Engines

Although radiation between clustered motors exists, the amount of resultant overheating of the motor will not be great unless the motors are arranged so that the combustion chambers or throats are very close. Close proximity of the expansion nozzles is not a problem because they are well below the limiting coating temperature.

(4). Heat Transfer to Vehicle

Radiation cooled motors may be required to operate in the vicinity of a portion of the vehicle which should absorb only a limited amount of radiant heat from the motor. Radiation shields, combined with high thermal conductivity heat sinks or insulators can reflect the radiation to space unless the motor is so completely surrounded by the vehicle that a separately cooled radiation shield is required.

(5). Advanced Nozzle Types

Radiation cooling of other motor configurations than a convergent-divergent nozzle would be seriously limited because almost all other configurations use a plug or similar structure to form the throat, and the shape factor for radiation to space from the plug throat is quite small. Some portions of these configurations could be radiation cooled, however.

(6). Meteoroids

Meteoroid penetration of thin coatings on refractory metals is a possibility. Erosion or penetration of the coating on exterior surfaces exposed only to space vacuum is not critical and the penetration of the interior chamber surface has a much reduced probability. Radiation cooled exit skirts of coated molybdenum have been run successfully for complete duty cycles with holes purposely drilled through the metal wall and coating.

4. Ablative Cooling

a. Cooling Limitations

For liquid engine application, the oriented silica fiber reinforced phenolics have consistently shown superior performance over other ablative materials as combustion chamber liners. This has been attributed to the very viscous molten silica film which forms on the charred surface during operation.

As a throat material, silica reinforced phenolics have shown considerable promise for the earth storable propellants at pressures up to 150 psia and throat diameters of 1 inch and larger. Actual throat erosion rates are sensitive to run time and propellant injector performance.

For a typical application, the char depth and hence the required thrust chamber wall thickness increases with burning time to the one-half power as shown by the experimental data in Figure 8. Char rates for transient ablation and for a non-receding or non-eroding liner surface are not very sensitive to flame temperature or chamber pressure. However, surface erosion at the nozzle throat and at high velocity flow conditions limits run times with the cryogenic and space storable high energy propellants.

b. Operational Limitations

(1). Restart Capability

There do not appear to be any limitations on the restart capability of properly designed ablative chambers, either in a vacuum or at sea level. The only limitation appears to be that if the chamber is restarted before it is allowed to cool completely, a weight penalty will be imposed in terms of additional char thickness required. It has been previously postulated, and verified experimentally, that for long off times, the additional charring that takes place on shutdown is offset by the time delay before charring proceeds on the succeeding run due to the greater refractory barrier imposed by the thickened char structure. The added char depth due to postrun charring has not been completely evaluated.

(2). Short Pulse Operation Capability

There are no apparent limitations to short pulse operation except for the weight penalties imposed by excessive charring under this type of operation. The considerations are similar to those mentioned under restart except that the material is never allowed to cool below its char temperature during the cycling period and the char continues at the same rate during the "off" condition. Under Marquardt testing this has doubled the char for a short pulse (50% duty cycle) over that which would have been sustained for a steady state firing of the same accumulated firing duration. The magnitude of this factor would vary with the pulse width, "on" time versus "off" time (percent duty cycle) and "off" time between series of cycling bursts. Residual thrust due to postrun charring of reinforced phenolic is shown in Figure 9 for the case of a 1/16 inch char and resultant gas release.

(3). Throttling Capabilities

There are no detrimental effects in the throttling of ablative engines except as it affects the efficiency of the ablative process. As the chamber pressure is throttled to a lower value, the lower efficiency of the ablative process at the lower heat flux (due to incomplete cracking of gaseous pyrolysis products) causes the char to proceed at about the same rate.

(4). Storage Limits

There are some storage effects with all resin systems since they all degrade to a degree in time when exposed to temperatures well below their char temperature. Presently considered phenolic systems have been the most widely evaluated under heat, vacuum, and ultraviolet radiation. It is estimated that about 10% of a phenolic will volatize in one year at 500°F under a hard vacuum.

(5). High g Operation

Little information is available on high g effect. However, it could be detrimental in displacing a molten reinforcement at the ablating surface especially, if the chamber is shut down under the application of a large g force.

(6). Meteoroids

The effects of meteoroid penetration on reinforced phenolics are not predictable at present. The thicker walls would appear to give greater resistance to penetration than the thin tubing or coated refractories.

(7). Space Radiation

Phenolic resin systems and others are adequately stable under space levels of radiation.

(8). Outside Wall Temperatures

The structural requirements of reinforced phenolics permit operation at exterior wall temperatures between 500° and 800°F without extra insulation.

5. Film Cooling

In film cooling, the fluid is introduced directly into the thrust chamber. This layer of fluid or gas then absorbs heat and thickens the effective boundary layer and reduces the heat flux to the thrust chamber surfaces.

Cooling films may be generated in several ways as follows:

1. Liquid fuel or oxidizer injected through wall slots or holes in the combustion chamber ahead of the critical nozzle area

2. Separate injection of propellant along the chamber walls from the propellant injector

3. Design of the injector to provide a fuel-rich, reacted gas mixture along the chamber walls

4. Evaporative heat sink of coolant discharging into the combustion chamber

Film cooling may be used effectively to protect the chamber walls in several ways as follows:

1. Reduction of the "adiabatic" wall temperature to a value below the material limiting temperature

2. A reduction in the heat flux to a wall which is also cooled by radiation, convection, or a heat sink

3. Maintaining a non-oxidizing gas adjacent to refractory surfaces otherwise capable of withstanding full combustion gas temperature, such as uncoated tungsten, tantalum, or various carbides

a. Cooling Limitations

There are no apparent limitations on cooling capability, time, or chamber pressure with either film or transpiration cooling. If one of the propellants (usually the fuel) or an inert fluid is used as a coolant at the nozzle throat, there is a performance penalty (I_{sp} loss) due to gas and temperature stratification. Figure 10 indicates that a typical performance loss due to film cooling is proportional to the quantity of coolant flow.

b. Operational Limitations

Pulsing and multiple starts may result in coolant waste due to a requirement to establish coolant flow prior to ignition and also from residual flow from coolant passages after shutdown. Plugging of cooling passages or transpiration media may be caused by thermal decomposition of coolant under cycling conditions.

6. Transpiration Cooling

Transpiration cooling may be thought of as a special case of film cooling and many of the same design considerations apply. The transpiration effect may be produced in several ways including the following:

1. Fuel forced through a porous wall

2. Water or other coolant delivered from a reservoir and pumped through a porous surface

3. A porous refractory slab filled with copper, lithium, subliming salts, etc., which are vaporized and discharged into the thrust chamber

This form of cooling is most applicable to one-shot, constant thrust engines due to the problems of flow control and shutdown effects.

7. Heat Sink Cooling

a. Cooling Limitations

Combustion chamber component temperatures may be held below structural limits while heat is being conducted away from the surface and absorbed in the chamber walls. The primary limitation on this concept is the run time available before a limiting surface temperature is reaced. Two limiting temperatures are encountered: First, the melting, subliming, or softening temperature at which the material would flow or erode rapidly, and second, the temperature at which the oxidation rate or reaction rate with the combustion gases would be excessive.

Promising heat sink materials are those which have high heat capacity, high thermal conductivity, high structural temperature limits, and compatibility with combustion gases. Pyrolytic graphite, isotropic graphite, and tungsten top the list for use with high temperature propellants. Oxidation is the critical problem with combustion gases containing CO_2 and H_2O. Graphite and tungsten surface coatings offer only a partial solution to this problem, since available coatings are limited to temperatures of less than 4000°F.

Surface temperature rise rates for isotropic and pyrolytic graphite in a combustion environment are shown in Figures 11 and 12. Temperatures of an insulated pyrolytic graphite insert in a 4 inch diameter nozzle throat, would be less than 3000°F for 200 seconds at 150 psia chamber pressure and 5000°F gas temperature. However, at more severe conditions such as 300 psia and 7000°F gas temperature, the 3000°F surface temperature would be reached in 10 seconds.

Theoretically, the run times for heat sink nozzles can be extended through the use of endothermic heat sink materials. These are materials such as subliming salts, lithium compounds, and low melting point metals capable of absorbing large amounts of heat through a phase change from an initial solid state. The endothermic materials may be impregnated into porous refractory wall materials or used to back up the walls as an insulator as well as a heat sink.

b. Operational Limitations

(1). Pulsing Operation

Inert heat sinks are best suited to low duty cycle pulsing operation. Indefinite run times can be achieved with limited radiation cooling. Endothermic heat sinks would not be applicable.

(2). Throttling

No limitation except total run time. Throttled operation increases available run time.

(3). Meteoroids

Heavy walled sections provide minimum effects due to meteoroid damage.

(4). Exterior Wall Temperatures

The structural limits of heat sink materials may permit operation at exterior wall temperatures above 4000°F. If environmental requirements do not permit this, available insulations can be used to reduce the exterior temperatures to less than 300°F and a minimum heat flux with some weight increase.

VI. PRELIMINARY THRUST CHAMBER WEIGHT ANALYSES

Selection of a cooling method from several which are applicable over the same required range of operating conditions may be made on the basis of thrust chamber weight. This section presents typical component weights for different cooling methods to facilitate this weight comparison. Injector and attachment flange weights are not included in this section.

A. Typical Thrust Chamber Configurations

The typical thrust chamber configuration lines used in these comparisons are shown in Figure 4 for a 40 to 1 exit nozzle expansion ratio. Combustion chamber contraction ratios (A_c/A_*) vary for different applications but in general they decrease at higher thrust levels whereas the ratio of thrust chamber volume to nozzle throat area (defined as L*) increases with thrust. For the purpose of making a weight comparison study, nominal values of contraction ratio are assumed to be between 4 and 2, and L* is assumed to vary as shown in Figure 13.

Nozzle thrust coefficient (C_F) varies with propellant, chamber pressure, and expansion ratio. But to provide a basis for weight comparison, a fixed value of 1.89 is assumed based on $A_e/A_* = 40$. The variation with propellant and expansion ratio is shown in Figure 14. For an evaluation of the effect of varying nozzle expansion ratio on weight and performance, C_F may be varied accordingly. Figure 15 presents a plot of engine throat diameter versus chamber pressure and thrust for use in the weight study based on the equation

$$F = C_F A_* P_c$$

The exit nozzle contour is assumed similar to the Rao contour with a length from the nozzle throat to the exit plane equal to 75% of the length of the equivalent 15° divergent cone. This length may be expressed by the equation

$$L_n = 1.35D^* \left[\left(\frac{A_e}{A_*} \right)^{1/2} - 1 \right]$$

which is plotted in Figure 16.

Thrust chamber and nozzle surface areas as a function of throat diameter contraction and expansion ratio are given in Figures 17 and 18.

Fairly detailed typical weight data are presented for regenerative, radiation, and ablatively cooled thrust chambers. For the purposes of a preliminary weight analysis, it may be postulated that the structural weights of dump cooled (open tube), film cooled, and transpiration cooled structures are the same as the weights of the regeneratively cooled thrust chamber. It is also postulated that the heat sink thrust chambers are equal in weight to the ablative thrust chambers. For the limited number of cases evaluated, these assumptions proved adequate, the choice would not be based primarily on a chamber weight comparison, especially for these latter cooling methods.

Examples of the use of these weight studies to make specific weight comparisons are shown in Figures 19, 20, and 21 for the cases of a long run throttling engine, a fixed total impulse engine of varying thrust and run time, and a minimum weight engine versus thrust and burn time. Details of these studies are presented in Section VIII.

B. Weights of Regeneratively Cooled Thrust Chambers

Due to the large number of variables involved in tube wall chamber design, it is difficult to illustrate trends in thrust chamber weight by use of a single curve. For this reason, the thrust chamber (Figure 22), excluding propellant injectors, were divided into a number of areas and the weight of each is presented on a separate curve. The separate areas of consideration were as follows:

1. Chamber reinforcement weight upstream from the throat (Figures 23 and 24)

2. Nozzle reinforcement downstream from the throat (Figure 25)

3. Coolant passage weight upstream from the throat (Figure 26)

4. Coolant passage weight downstream from the throat (Figure 27)

5. Coolant manifold weights for N_2H_4 and H_2 fluids (Figures 28 and 29)

6. Coolant weight in tube passages upstream from the throat (Figure 30)

7. Coolant weight in tube passages downstream from the throat (Figure 31)

8. Coolant weight in manifolds for N_2H_4 and H_2 fluids (Figures 32 and 33)

These eleven graphs (Figures 23 through 33) illustrate the effect of chamber pressure, thrust, throat area, expansion and contraction ratio, minimum gage requirements, and coolant density of the weights of items comprising a regenerative cooled thrust chamber. Metal density and strength correspond to an alloy such as Type 321 stainless steel.

Use of the above eleven graphs allows flexibility in determining the effect of any single or combination of parameters on chamber weight. The ordinates of all the graphs are plotted in terms of Weight/Throat area. A total chamber weight is arrived at by the addition of all applicable individual factors and then multiplying the total by the throat area.

Predicted weights for several thrust chambers of different chamber pressure, expansion ratio, and throat area are given in Figure 34 for the O_2/H_2 propellant combination. This is representative of the more specific types of results that can be obtained from the set of weight curves.

Weight information as presented in and determined from the graphs in this section is not intended to represent the shelf weight of regeneratively cooled thrust chambers, since actual delivery weight is a strong function of specific details of size and application. However, the accuracy of the curves should be within 10 to 15 percent.

C. Weights of Radiation Cooled Thrust Chambers

The weights of radiation cooled motors of the configuration shown in Figures 35 and 36 were based on the following assumptions:

1. Motor wall temperatures for the propellant system N_2O_4/0.5 N_2H_4-0.5 UDMH with 95 percent C* efficiency

2. Wall emissivity factor = 0.72

3. Effective shape factor = 1.0 in combustion chamber

4. Material selection above 2000°F: 90% tantalum-10% tungsten using tensile strength for 1 percent creep in 10 minutes

5. Material selection below 2000°F: Haynes 25 alloy using tensile strength for 0.2 percent yield

6. Minimum wall thickness in all cases = 0.020 inch

The weight of radiation cooled motors using 90% tantalum-10% tungsten throughout is shown in Figure 35. The weight of motors using 90 Ta - 10W in the chamber and throat and Haynes 25 alloy in the expansion nozzle where metal temperatures are below 2000°F is shown in Figure 36. The maximum wall temperature for many of the combinations of chamber pressure and thrust indicated in Figures 35 and 36 exceeds the 3300°F limit of coatings currently available, and hence are not feasible from an oxidation standpoint. Chamber weights for O_2/H_2 propellants would be approximately equal to those shown here with the same limit on coating temperatures.

Weight estimations for radiation cooled expansion skirts for area ratios greater than 40:1 are facilitated by the curve of nozzle surface areas plotted in Figure 18. The areas shown are exact for a nozzle contoured for a 40:1 area ratio, but are approximate for alternate expansions.

D. Weights of Ablative Thrust Chambers

Weights for typical ablative thrust chambers as a function of run time and size presented in Figures 37 through 41 were based on the following assumptions:

1. Material weight is based on silica reinforced phenolic with a density of 0.0625 lb/cu in.

2. Steady state char depth data is based on firing data in the 25 to 2000 pound thrust range taken from References 3 to 6 and recent unpublished Marquardt data. A design curve for weight analysis is shown in Figure 8 for the combustion chamber and throat region. For times less than 60 seconds, the design curve gives a more conservative wall thickness.

3. Wall thicknesses required in the exit nozzle and expansion skirt section are reduced due to lower heat fluxes and re-radiation from the inner nozzle surfaces. Wall thickness scaling factors shown in Figure 42 are based on altitude firings of 25 and 100 pound thrust ablative chambers.

4. Char rate is assumed to be independent of chamber pressure. Within the range of experimental data, no direct effect on char rate has been observed for chamber pressures of from 50 to 500 psia. Throat erosion rates, however, are known to be a function of chamber pressure but have not been correlated as such.

5. Char rate is assumed to decrease for small chambers where the chamber radius approaches the wall thickness (Reference 7).

6. The weight contribution of the structural pressure containing shell of the thrust chamber is assumed to be the same for a metal or a resin bonded fiberglass design on the basis of similar strength-to-weight requirements and the small fraction of chamber weight contributed by the outer shell.

7. The separate weight of a nozzle throat insert is not included. A coated graphite insert would have nearly the same density (0.067 lb/cu in.) as the silica phenolic insert. Some additional wall thickness would be required under the insert for increased char depth.

8. Chamber weights for other propellants would be the same for the non-eroding components.

9. For ablative exit nozzle skirts of less than 40:1 expansion ratio, the curves of Figure 41 may be used to calculate weights for each section of the thrust chamber for any run time.

VII. PROPULSION PERFORMANCE PENALTIES

A. I_{sp} Losses Due to Film and Transpiration Cooling

If a film of liquid or gas flows through a rocket nozzle throat at a temperature different than that of the main bulk of exhaust gas, the net thrust of the engine will be less than that which would result if the gases had been thoroughly mixed with the same overall total enthalpy. This gas stratification effect is independent of the effective chemical combustion efficiency. The analytical evaluation of this phenomena is presented in Appendix B of Volume II. The magnitude of this effect on I_{sp} is presented for various film temperatures and film thicknesses in Figure 43.

An additional I_{sp} loss may be incurred due to the operation at propellant mixture ratios other than optimum in order to insure sufficient propellant as film coolant. Ideally, for a 5% hydrogen film coolant flow, this loss could be less than 1%. The stratification loss could vary from 2 to 5% depending upon the effective film temperature.

Experimental data as shown in Figure 10 (from Reference 8) confirm a performance loss approximately equal to the percentage of coolant flow. For preliminary design, this is the recommended value to use.

There is some recent experimental evidence that I_{sp} losses may be incurred with an ablative thrust chamber due to the transpiration effect of the ablative material. However, no numbers are available to evaluate the separate effects of shear force losses, changes in contour, or throat erosion as well as the transpiration film effect. A typical gas generation rate from the thermal degradation of an ablative liner at normal char rates is less than 1/10 of 1 percent of the propellant flow, so that this should be a negligible loss.

B. Thrust and I_{sp} Changes Due to Throat Erosion

Nozzle throat erosion, if controlled and predictable, could be acceptable in some engine applications. The effect on thrust, propellant flow rate, and I_{sp} have been calculated for throat enlargements up to 25%. For fixed area propellant injectors and fixed propellant supply pressure, engine thrust would increase while decreasing in I_{sp} performance. An I_{sp} loss of only 0.5% would be incurred for as much as 10% increase in throat area. This effect is shown in Figure 44 as a function of throat area increase and propellant injection pressure ratio for a 40:1 expansion thrust chamber. The further assumption has been made that there are no aerodynamic losses due to distortions in the nozzle contour.

C. Heat Losses and Pressure Losses

Heat losses from combustion gases to thrust chamber walls and the pumping energy required to overcome pressure losses in propellant and coolant liner result in a loss in impulse efficiency (I_{sp} loss) equal to one-half of the ratio of the energy loss to the total gas enthalpy. The theoretical relationships are worked out in Appendix B to Volume II of this report.

In a typical 2000 pound thrust radiation cooled engine, the total heat flux lost through the combustion chamber walls would be 72 Btu/sec. This is approximately 0.6% of the total gas flow enthalpy. Hence, the I_{sp} loss due to heat transfer would be 0.3%.

D. Residual Thrust in Ablative Engines

After an ablative thrust chamber has been running for several seconds and stopped, the heat stored in the charred phenolic and silica reinforcement must soak into the virgin material. Thermal degradation of the virgin material will continue to occur until the mean temperature of the char is reduced to near 500°F. Postrun charring of 0.062 to 0.25 inch of virgin phenolic may be calculated depending upon the char depth at shut down. However, limited experimental data on postrun temperatures indicate that a somewhat thinner post char thickness actually develops.

The weight of gas generated due to charring is approximately 15% by weight of the ablative material which is charred. If the gas released during the postchar period, which may be as long as 100 seconds, is heated in the chamber to an average of 1100°F, a residual postrun impulse may be calculated, as shown in Figure 9, as a function of thrust and chamber pressure. The curves show a total postrun impulse for 0.062 inch char in a 100 pound thrust engine is 3 lbf-sec. This is equivalent to a 30 millisecond pulse width which is greater than the desired minimum typical pulse widths shown in Figure 45. However, if pulse firing were the normal mode of operation, less severe temperature gradients in the walls would greatly reduce postrun charring.

In Table I (mission requirements), a typical value of large engine thrust to spacecraft mass is 1.0 and a typical value of impulse cutoff accuracy is 1.0 lbf-sec per pound of spacecraft mass, hence an allowable impulse of 1.0 lbf-sec per lbf of engine thrust may be typical. Values of residual impulse shown in Figure 9 are all below 0.1 and 0.01 lbf-sec per lbf. However, within the probable ranges of these variables, postrun charring may be a design consideration.

E. Optimum Exit Nozzle Expansion Ratio Versus Engine Performance, Weight, and Size

The performance gain in I_{sp} associated with increasing exit nozzle expansion ratios is attained at the expense of increased exit diameter, increased nozzle length, and increased nozzle weight. The attainment is also dependent on whether the flow achieves frozen or shifting equilibrium.

The potential gain in performance (I_{sp}) for different propellants is shown in Figure 46 for expansion ratios of from 15 to 800 with shifting equilibrium.

The weight penalty associated with large expansion ratios consists of

1. The weight of nozzle skirt, which may be radiation cooled at large expansion ratios

2. Increased structure weight associated with increased supporting loads

3. Increased structure weight of surrounding structure due to increased engine diameter and length

As an alternate concept, a net performance gain within a fixed engine envelope and fixed thrust may be possible by using very high chamber pressures with a very large expansion ratio. The increased severity of the cooling problem may be approached by the use of film and transpiration cooling. There may be a net gain if the coolant losses can be minimized and shifting equilibrium performance approached. An example of this trade-off is given in the following table which considers the case of increasing chamber pressure from 50 to 1000 psia and A_e/A_* from 40 to 800 to provide a constant exit diameter. The greatest potential gain is with the OF_2/B_2H_6 propellants if the cooling problem can be solved.

Propellant	O/F	Max. I_{sp} at 50 psi $A_e/A_* = 40$	Max. I_{sp} at 1000 psi $A_e/A_* = 800$	Percent Increase I_{sp}
OF_2/H_2	7.0	473 sec	509 sec	7.5
OF_2/B_2H_6	4.0	430	494	15.
O_2/H_2	5.0	453	494	9.

VIII. DESIGN STUDIES AND FACTORS AFFECTING FINAL CHOICE OF COOLING METHOD

A. Design Studies

To illustrate the practical application of the cooling technique selection procedures presented in this report, four specific propulsion requirements are postulated and evaluated for applicable and best thrust chamber designs.

1. Example 1: Variable Thrust, Deep Space, Liquid Rocket Engine

a. Specification of Propulsion System

(1). Engine Purpose

The purpose of this engine is deep space, mid-course propulsion, to start and operate in deep space environment only.

(2). Propellants - Earth Storable

This specification might include the choice of such oxidizers as ClF_3, N_2O_4 or mixed oxides of nitrogen. Although the use of ClF_3 results in slightly higher flame temperature, its use would mainly limit the use of coated refractories in a radiation cooled thrust chamber. The choice of fuel for maximum performance within the state of the art would be one of the amines such as N_2H_4, UDMH, or a blend. From the standpoint of regenerative cooling, the best choice is hydrazine with an additive such as EDA. Another common blend suitable for radiation and ablative cooled engines is the Aerozine-50 (0.5 N_2H_4-0.5 UDMH). Fuels such as MMH are similar to Aerozine-50 with respect to cooling capabilities and are not considered in detail in this report.

For the purpose of this design study, the following propellants are considered:

$$N_2O_4/(N_2H_4 + 10\%\ \text{EDA})$$

$$N_2O_4/0.5\ N_2H_4 - 0.5\ \text{UDMH}$$

In performance and flame temperature, these propellants are quite close. For the regeneratively cooled design, the N_2H_4 + 10% EDA is required.

The mixture ratio is chosen to give maximum I_{sp}. Performance loss at off-mixture ratios is not compensated by the resulting lower flame temperatures, and more than one cooling technique is feasible without compromising performance.

Postulated I_{sp} (theoretical) $\sim$ 338 seconds

I_{sp} (delivered) $\sim$ 300 seconds

(Based on 89% efficiency and an expansion ratio of 40:1.) Higher efficiencies should be attainable for this application. This value is used primarily to calculate propellant weight and to evaluate equivalent propellant weight gains or losses due to changes in nozzle expansion area.

(3). Propulsion Specification

1. Initial spacecraft weight = 4000 lbm
2. Three successive duty cycles for the single engine.
   a. F = 500 lbf, ΔV = 600 fps, 4 starts
   b. F = 2000 lbf, ΔV = 10,000 fps, 2 starts
   c. F = 500 lbf, ΔV = 900 fps, 2 starts
3. Total coast time in space = 240 days
4. No other limitations on system design at this point.

Using the following equation relating velocity change, specific impulse, and spacecraft weight change, the burning times and propellant weights required for the above propulsion cycles were calculated.

$$\Delta V = g\, I_{sp}\, \mathrm{Ln} \left[\frac{W_{initial}}{W_{final}}\right]$$

These calculations provide the following propulsion system specifications.

Thrust	ΔV	Run Time	Starts	Propellant
500 lbf	600 fps	180 seconds	4	300 lbm
2000 lbf	10,000 fps	455 seconds	2	3028 lbm
500 lbf	900 fps	91 seconds	2	152 lbm
	Totals	726 seconds	8	3480 lbm

(4). Engine Configuration

A conventional convergent-divergent engine configuration is chosen as the easiest to cool. If another configuration appears to have some advantage from a structural consideration, it may be compared with the results of this study.

For the purpose of the cooling method study, a nozzle expansion A_e/A_t = 40 is chosen. The weight and performance trade-off in going to a larger or smaller value may be made by means of the following table based on curves of I_{sp} versus A_e/A_* and surface areas.

Assumptions: A_e/A_* = 40:1 used as basis of comparison

P_c = 150 psi

Extension skirt made of 0.030 stainless steel

A_e/A_*	C_F	I_{sp}	Equiv. Wt. Propellant	Skirt Wt. Δ lb	Exit Diameter	Δ Length (ins.)	Interstage Structure Wt.
30	1.87	298	+ 35 lb	- 3.1 lb	16.4 in.	-5	?
40	1.89	300	0	0	19.0 in.	0	?
100	1.93	306	- 73 lb	+ 9.3 lb	30.0 in.	+14	?

The combustion chamber geometry may be fixed finally from combustion considerations, but from a cooling area and chamber weight standpoint, the larger nozzle contraction ratios for a given L* or combustion volume result in a somewhat lighter structure. For the cooling studies, a representative curve of L* is used to select chamber volume, and the contraction ratio is selected on the basis of the cooling method as being 4. The general lines used are presented in Figure 4.

The single engine is located at the aft end of the spacecraft with no inherent envelope or size limitation indicated.

b. General Applicability Screening

Scanning the screening charts and reviewing the more critical factors relative to run time, restarts, engine envelope, and propellant choice, the following cooling concepts appear to be applicable:

1. Regenerative (N_2H_4 + EDA) (See Figure 5)
2. Radiative
3. Ablative
4. Film
5. Combinations of the above

With respect to film cooling -- there is an inherent complexity and performance penalty that puts it at a disadvantage when radiation and regenerative cooling are both possible. Hence, it is considered non-competitive in this problem. Furthermore, where the propellant weight is large compared to the chamber weight as in this case, performance penalties are even more critical. A performance penalty of 2% I_{sp}, in terms of propellant requirement, would cost more than the weight of the thrust chamber.

c. Preliminary Design Comparison

(1). Weight Analysis

Preliminary design layouts and structural weights may be calculated using curves such as the following:

1. Throat diameter versus thrust and pressure (Figure 15)

2. Exit nozzle surface area versus expansion ratio and throat diameter (Figure 18)

3. Combustion chamber surface area versus throat diameter and contraction ratio (Figure 17)

4. Expansion nozzle length versus throat diameter and expansion ratio (Figure 16)

For each cooling method there are plots of typical structure requirements as a function of chamber size and chamber pressure either in Volume I or II. Some of these are:

1. Combustion chamber reinforcement and passage weight versus throat area for regenerative cooling

2. Char depth versus run time for ablative chambers

3. Equilibrium wall temperatures for different radiation cooled operating conditions

4. Ablative thrust chamber weight versus run time, chamber pressure and thrust (Figure 40)

5. Radiation cooled thrust chamber weight versus thrust and chamber pressure (Figure 36)

6. Throttling limits for regeneratively cooled chambers

7. Coolable expansion ratios for regeneratively cooled designs

8. Structural material capabilities

9. Regeneratively cooled thrust chamber weight versus throat diameter (Figure 34)

Preliminary thrust chamber designs and thrust chamber weights may be obtained using the above graphs. These weights may be calculated for a range of chamber pressures as shown in Figure 19. Comments on the designs represented by these weights are given below.

(2). Regenerative Cooling

1. Regenerative cooling is possible with (N_2H_4 + EDA) but not with Aerozine-50.

2. Allowable chamber pressure ranges for 4:1 throttling and minimum passage size of 0.062 inch are given below (Reference Section III-A of Volume II).

Thrust	$P_{c_{max}}$	$P_{c_{min}}$
500 lbf	60 psia	30 psia
2000 lbf	240 psia	120 psia

3. Cooled expansion ratio = 10:1. Assume radiation cooled refractory metal skirt from $A_e/A_* = 10$ to 40.

4. Propellant supply pressure variation with fixed orifice injectors at

$$P_c = 60 \text{ psi}, \ P_{sup} = 95 \text{ psia}$$

$$P_c = 240 \text{ psi}, \ P_{sup} = 700 \text{ psia}$$

For a two-thrust level design, a variable area injector could be used to reduce the propellant supply pressure variation.

5. Design considerations to be evaluated in more complete design study include:

Meteoroid damage

Zero gravity effects

Freezing of propellant in cooling passages during deep space coasting

Cut-off impulse accuracy

6. Weights in Figure 19 include weights of

   Chamber reinforcement

   Cooling passages

   Manifolds

   Fuel in cooling passages and manifolds

   Radiation cooled, 0.020 inch columbium skirt from $A_e/A_* = 10$ to 40

(3). Radiation Cooling

1. The maximum allowable theoretical equilibrium chamber wall temperature = 3300°F with wall emissivity = 0.72. This limits chamber pressure to 50 psia (Volume II).

2. The effects of internal radiation and axial heat conduction are considered minor.

3. The chamber material is silicide coated 90% Ta-10% W alloy with a minimum gage of 0.020 inch. A less dense metal such as stainless steel can be used in the expansion skirt at area ratios where the equilibrium wall temperature drops below 2000°F (Figure 36).

(4). Ablative Cooling

1. The char rate is independent of chamber pressure over the range of interest.

2. A duty cycle requiring several closely spaced firings increases the char rate over steady state or widely spaced firings. This chamber is designed for 1000 seconds of steady state firing.

3. The chamber material is silica fiber (oriented cloth) reinforced phenolic.

4. The chamber pressure stresses are taken by either metal can or fiber glass wrap (Assumed equivalent for weight study).

5. The use of a hard throat insert may be required depending on the chamber pressure and injector design. Maintaining the throat becomes more difficult at the higher chamber pressures. The choice has small effect on chamber weight.

6. Figure 19 shows two ablative chamber designs. The use of ablative material all the way to 40:1 may be required if there are limits on the outer wall temperature. If the skirt is free to radiate, a refractory metal skirt may be used beyond the area ratio producing equilibrium wall temperatures below 3000°F (taken as 10:1 for design study).

d. Factors Affecting Final Choice

In this particular problem, the long run time of 726 seconds and the 4:1 throttling range are the most demanding requirements. The lightest chamber design shown by Figure 19 is the regeneratively cooled chamber operating at 250 psia at the 2000 pound thrust level. Two factors which affect the choice of the regenerative cooling design are the requirement for either a high propellant supply pressure or a variable area injector, and the requirement that the cooling passages be purged after shut down.

The second choice could be either the radiation cooled or the ablative engine with a radiation cooled skirt. The choice may be based on a system study which would include the engine envelope restrictions and the propellant supply system weights.

Meteoroid effects during the 240 day coast may have an effect on chamber design choice if more data were available.

2. Example 2: Constant Thrust, Oxygen-Hydrogen Fueled Space Engine

a. Specification of Propulsion System

(1). Engine Purpose

This engine study was made to demonstrate the variation of cooling method applicability with thrust and run time for a minimum weight thrust chamber. The results are plotted in Figure 21.

(2). Propellants

Liquid oxygen-hydrogen

(3). Propulsion Specification

1. Engine thrust = Constant

2. Thrust range = 20 to 10,000 lbf

3. Engine burning time = 3 to 1000 seconds

4. Number of starts = 1

5. Thrust chamber pressures (P_c maximum)

Radiation cooled, P_c = 50 psi

Ablative cooled, P_c = 150 psia

Heat sink, P_c = 150 psia

Regeneratively cooled, P_c = See Figure 5

(4). Environmental and Operational Requirements

1. Engine free to radiate

2. No envelope restrictions

3. Convergent-divergent nozzle, $C_r = 4$, $A_e/A_* = 40$

b. General Applicability

This study was conducted for the four cooling methods indicated above.

c. Weight Study

Radiation cooled chamber weights were based on Figure 36 for 50 psia chamber pressure. Weights assumed independent of run time for times less than 1000 seconds.

Ablative chamber weights were based on Figures 37, 38, and 39 at 150 psia for a reinforced phenolic nozzle throat design. A nozzle throat insert of coated graphite was assumed for small thrust engines (below 500 pounds).

The heat sink thrust chamber was assumed to be of graphite with weights equal to or lighter than ablative chambers for short run times. Heat sink was applied where transient throat temperatures fell below 2500°F.

Regeneratively cooled thrust chamber weights were calculated for the minimum thrust versus pressure engine sizes shown in Figure 5.

d. Discussion of Results

This study was made for the purpose of defining the general areas of applicability. Figure 21 shows that, on a weight basis, the best applications for the cooling methods shown are:

1. Radiation cooling - Low thrust, long run times

2. Ablative cooling - Low thrust, run times from 10 to 300 seconds

3. Regenerative cooling - High thrust, medium to long run times

4. Heat sink - Short run times

3. Example 3: Constant Total Impulse Engine with Firing Time and Thrust as Variable Parameters

In space, some maneuvers such as orbital changes require a particular total impulse and are not sensitive to firing time (within limits). This study indicates (Figure 20) that although both ablative and radiation cooled thrust chamber weights can be reduced by increasing the run time and decreasing thrust, ablative thrust chamber weights are affected by the thicker walls required for the longer run times. Thus, there is, in this study, a weight crossover point at 200 seconds run time with the radiation cooled chamber being the lightest for the longer run times.

The weights in these curves were taken from Figures 35, 37, 38, 39, and 40.

4. Example 4: Mars or Venus Orbital Flight

a. Specification of Propulsion System

(1). Engine Purpose - Deep space, mid-course or orbital braking propulsion.

(2). Propellants - Space storable OF_2/B_2H_6

(3). Propulsion Specification

1. Thrust = 4000 lbf (constant)

2. Run time = 300 seconds (total)

3. Number of restarts = 4

4. Minimum run time = 10 seconds
   Maximum run time = 300 seconds

5. Number of engines = 1

6. Specific impulse = 400 seconds
   Gas temperature = (6500° to 7500°F)

7. Chamber pressure = Fixed by minimum system weight and reliable operation

(4). Environmental and Operational Requirements

1. Engine location = Free to radiate.
   Equilibrium soak temperatures during coasting = 260° to +100°F

2. Engine envelope limitations = None

3. Engine configuration = Weight study based on Figure 4, C-D nozzle, $A_e/A_* = 40$, $C_r = 4.0$, $D_* = 4.17$ inch

4. Oxidizer/Fuel ratio = 4.0

5. Storage time in space = 250 days

b. Applicable Cooling Techniques (See Table II)

(1). Radiation Cooling

From Figure 36 in Volume II, which presents radiation cooled wall temperatures for OF_2/B_2H_6 at P_c = 20 psi, it can be seen that the nozzle throat temperature would be 3500°F at a radiation factor of 1.2 and would drop to 2000°F at an area ratio of 10. The compatibility of coated or uncoated refractory metals at 3500°F with these combustion gases has not been established. Hence, this is a tentative possibility at best.

(2). Heat Sink - (See Figures 11 and 12)

At 150 psi chamber pressure (h = 550 Btu/hr ft^2 °F), a typical heat sink throat temperature using an edge oriented pyrolytic graphite heat sink would be 4500°F in 300 seconds. At 600 psi, the surface temperature would approach 6200°F in 300 seconds. The rates of erosion and oxidation of pyrolytic graphite for these gas environment conditions are unknown but the cooling concept for a compatible combustion gas is structurally feasible.

(3). Ablative Cooling

Ablative materials could be considered applicable to a part of the thrust chamber and exit cone but not to the throat. Even in the combustion chamber, run times of 300 seconds would doubtless cause considerable surface erosion depending on chamber pressure. Experimental data are very limited.

(4). Film and Transpiration Cooling

Figure 47 compares three analytical approaches to film cooling the exit nozzle with B_2H_6 based on References 9, 10, and 11. (Discussed in Volume II, Section III.) Straight liquid film cooling (Case I) is obviously not practical. Gas film cooling (Case II) also requires a fairly large fraction of the propellant flow to cool to an exit area ratio of 10. However, if the results for Case III could be achieved in practice, as little as 3% of the total propellant flow would be required to cool the nozzle from ahead of the throat ($A_e/A_* = 1.5$) to downstream from the throat ($A_e/A_* = 10$). With transpiration cooling, the predicted performance is about the same, or about 3% of the fuel required as coolant. Figure in Volume II, indicates that the amount of coolant required in terms of percentage of propellant decreases with increasing chamber pressure.

c. Preliminary Weight Analysis

(1). Thrust Chamber Configuration

The thrust chamber lines shown in Figure 4 were used for weight comparisons.

(2). Propellant Weight

Total impulse at 4000 lbf and 300 seconds run time,
I_t = 1,200,000 lbf-second

Total propellant weight at I_{sp} = 400 seconds

W_p = 3,000 lbm

(3). Weight Comparison

Rough weight comparison based on available curves.

Radiation cooled, Figure 36 at 20 psi		W_s = 110 pounds
Ablative cooled, Figure 38 at 150 psi (with zero erosion) 300 sec		W_s = 93 pounds
Film cooled, +3% coolant	Figure 34 at 150 psi	W_s = 35 pounds W_c = 90 pounds
	Total	125 pounds

d. Factors Affecting Final Choice

Radiation cooling, even at 20 psia, appears to be marginal at best. The I_{sp} performance at 20 psia compared to 150 psia is lower by 2.5%. The thrust chamber size at 20 psia would be about three times the length and diameter of the 150 psia engine. Hence, there would be no weight advantage with radiation cooling.

The heat sink thrust chamber also appears marginal for 300 seconds using pyrolytic graphite and would doubtless weigh more than 90 pounds or more than 3% of the propellant weight.

The optimum design approach recommended is the use of film or transpiration cooling in combination with a pyrolytic graphite heat sink throat insert and graphite chamber liners upstream and downstream from the throat protected from oxidation by a minimum of protective inert film. If the inert film protection concept can be developed for application to thrust chamber pressures of 150 psi and above, the use of higher nozzle expansion ratios and smaller thrust chambers will provide optimum propulsion system performance.

B. Combined Cooling Techniques and Advanced Concepts

The foregoing sections have presented the applicability and limitations of individual cooling techniques. For many propulsion requirements, one of several cooling techniques may be used so that an optimum design may be selected on the basis of weight, complexity, or similar factors. However, there are several propellant systems for which the cooling requirements are of such severity that no completely satisfactory cooling technique has yet been developed.

Conditions which give rise to these severe environments are the use of fluorine based oxidizers such as OF_2, F_2, and ClF_3 in combination with fuels containing such metals as boron, aluminum, beryllium, and lithium. These propellants give combustion gas temperatures in the 6000° to 8000°F range. The severity of the combustion environment is further increased with increased chamber pressures. Furthermore, the combustion products are usually highly erosive and corrosive on the available refractory metals and carbides.

Throat heat fluxes fall in the 15 to 25 Btu/in.^{2} second range at chamber pressures of 600 psia. At these conditions, the very best inert heat sinks would reach temperatures of 5000°F in less than 20 seconds. Likewise, the other cooling techniques which do not involve a performance loss, such as regenerative, ablative, and radiation cooling will not do the job alone. Therefore, some form of film or transpiration cooling is required.

If film or transpiration cooling is required, then the objective of the design would be to minimize the coolant flow required and the attendant performance penalty (in terms of extra propellant or coolant weight required). Based on theory, there is a minimum coolant requirement which is based on the surface area to be cooled and the wall temperature. Therefore, the cooled surfaces should operate at the hottest possible temperatures at the nozzle throat consistent with structural integrity. Materials with the highest temperature capabilities are the graphites, tungsten, and the carbides of hafnium and tantalum. Structurally, graphite and tungsten are capable of operation above 5000°F. The structural capability of the carbides has not been demonstrated. However, all of these materials are subject to oxidation and erosion by the combustion gases even at 5000°F. Therefore, they must be both cooled and protected. Theoretically, this can be done with an injected film of inert fluid.

Most advanced cooling studies now in progress (References 5, 12, 13, and 14) are related to ways of generating this coolant film either on a transient basis or by providing a controlled steady state coolant film supply. An excellent review of work being done on materials and advanced cooling techniques for solid propellant motors is presented in Reference 13. Development problems lie in the areas of refractory material formulation, nozzle design and fabrication, coolant selection, and supply techniques. Particular problems include passage plugging by coolant or combustion products, coolant distribution, starting and shut down phenomena, limit on run time and thrust variability, and thermal expansion and sealing provisions.

Advanced combined cooling concepts which have shown promise but so far have been demonstrated only for limited run times include the following:

1. Porous refractories impregnated with lower melting metals or endothermic solids such as a subliming salt (Reference 14)

2. Porous throat inserts backed by a reservoir of endothermic heat sink material which absorbs heat in gasification. The gas flows into the chamber through the porous surface, providing a transpiration cooling effect (Reference 5)

3. Sacrificial inserts ahead of a throat insert (Reference 14)

4. Coolant in a liquid or gas reservoir which is pumped to cool the nozzle

5. A liquid metal reservoir to supply convective coolant to the back side of thin wall refractory metal nozzle

6. A radiation cooled heat sink of pyrolytic graphite

7. A film cooled heat sink to extend the inert heat sink running time with minimum performance penalty

8. A film cooled convective nozzle with coolant injected ahead of throat after being used to cool the throat convectively

9. A convectively cooled combustion chamber with coolant dumped into the chamber just ahead of the nozzle throat

The limitations of these cooling concepts have not been established, Continued research is required in the development of refractory materials, in the development of optimum film and transpiration coolant supply systems, and in experimentally defining the actual combustion environments.

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TABLE I

SUMMARY OF SPACECRAFT MISSIONS AND PROPULSION SYSTEM REQUIREMENTS

	Basic Mission Requirements								Representative System Characteristics								Possible Space Engine Configuration						
									Small Payload (Centaur or Saturn Launch)				Large Payload (Saturn or Nova Launch)										
Space Mission and Maneuver	Range of Ideal ΔV Requirements (fps)	Range of Desirable F / M_o (lb_f / lb_m)	Allowable Impulse Cutoff Accuracy $\Delta I_t / M_f$ (lb_f-sec/lb_m)	Required Thrust Variability	Restart Requirements	Typical Minimum Service Life (sec)	Thrust Program	Storability Requirements (Max.)	M_{P_L} (lb_m)	M_o (lb_m)	F (lb_f)	I_t (lb_f - sec)	M_{P_L} (lb_m)	M_o (lb_m)	F (lb_f)	I_t (lb_f - sec)	Thrust Required for Largest Nova H-8 Payload (lb_f)	Type Engine Used For Mission	Engine Location	Engine Attachment	Number of Engines	Thrust Level percent/ per type Main/Vernier	Mode of Thrust Correction
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
I. 300 N. Mi. Orbit																							
A. Orbital Correction																							
1. Orbital Perturbation	100 to 1,000	0.05 to 1.0	1.5	None	Multiple	240	1/3	Months	7100	8000	1000	0.24×10^6	36,000	40,000	5,000	1.2×10^6	60,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
2. Eccentricity Control	100 to 5,000	0.05 to 1.0	2.5	None	None	60	1	Months	7100	8000	4000	0.24×10^6	36,000	40,000	20,000	1.2×10^6	230,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
3. Orbital Plane Change	200 to 19,000	0.15 to 2.0	1.0	None	None	60	1	Months	7100	8000	4000	0.24×10^6	36,000	40,000	20,000	1.2×10^6	230,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
4. Orbital Altitude Variation	200 to 14,000	0.1 to 2.0	1.0	None	1 to 2	60	1/3	Months	7100	8000	4000	0.24×10^6	36,000	40,000	20,000	1.2×10^6	230,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
5. Orbital Epoch Change	100 to 20,000	0.1 to 2.0	0.5	None	1 to 2	60	1/3	Months	7100	8000	4000	0.24×10^6	36,000	40,000	20,000	1.2×10^6	230,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
6. Corr. of Injection Error	50 to 1,000	0.01 to 0.5	0.03	None	1 to 2	124	1/3	5 Days	7500	8000	1000	0.12×10^6	37,600	40,000	5,000	0.62×10^6	60,000	M	E	F	1 - M	100 - M	A
B. Orbital Rendezvous																							
1. Nominal Injection Errors	50 to 1,000	0.01 to 1.5	0.15 to 0.5	100 : 1	0 to 2	38	2/3	5 Days	7900	8500	4250	0.16×10^6	42,000	45,000	22,500	0.84×10^6	230,000	M	E	F	2 - M	100 - M	A
2. Dog-leg Maneuver	500 to 18,000	0.01 to 3.0	0.15 to 0.5	1,000 : 1	1 to 2	85	2/3	5 Days	3700	8500	25,500	1.3×10^6	20,700	45,000	135,000	7×10^6	1.4×10^6	M / V	M - E / V - E	M - F / V - G	4 - M / 4 - V	90 - M / 10 V	V
3. Emergency Rendezvous	1,000 to 25,000	0.01 to 3.0	0.15 to 0.5	1,000 : 1	1 to 2	105	2/3	1 Day	2000	4650	14,000	1×10^6	10,000	23,300	70,000	5.1×10^6	1.4×10^6	M / V	M - E / V - E	M - F / V - G	4 - M / 4 - V	90 - M / 10 V	V
II. 24 - Hour Orbit (19,310 N. Mi.)																							
A. Correction of Injection Errors	100 to 450	0.0004 to 0.2	2.0 to 15	None	Multiple	- -	1/3	1 Month	5650	6000	60	90,000	26,400	28,000	200	420,000	1,700	A	E	F	1 - A	100 - A	A
B. Station Keeping	50 to 130	0.0002 to 0.02	0.55 to 3.25	None	Multiple	- -	1/3	2 Years	5850	6000	40	19,500	27,300	28,000	150	91,000	1,400	A	E	F	2 - A	100 - A	A
C. Attitude Control	- -	- -	- -	None	Multiple	- -	1/3	2 Years	5850	6000	1	15,600	27,300	28,000	10	84,400	100	A	E	F	12 - A	100 - A	A
III. Lunar Flights																							
A. Trajectory Correction																							
1. Midcourse - Flt.	25 to 250	0.025 to 0.25	0.02	None	3	62	1/3	3 Days	2470	2500	125	7,750	98,500	100,000	5,000	0.31×10^6	5,000	M	E	F	2 - M	100 - M	A
2. Terminal - Flt.	25 to 500	0.020 to 0.5	0.03	None	2	62	1/3	3 Days	2470	2500	125	7,750	98,500	100,000	5,000	0.31×10^6	5,000	M	E	F	2 - M	100 - M	A
3. Terminal - Return	50 to 500	0.015 to 1.0	0.15 to 0.30	None	3	62	1/3	Months	490	500	25	1,550	19,800	20,000	1,000	62,000	1,000	M	E	F	2 - M	100 - M	A
B. Lunar Orbit	2,000 to 5,500	1.0 to 2.0	0.15 to 1.0	None	None	110	1	3 Days	1550	2500	2500	0.26×10^6	72,500	100,000	100,000	11×10^6	100,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
C. Lunar Landing																							
1. Direct	9,800 to 9,900	1.0	- -	6 : 1	None	330	2	3 Days	700	2300	2300	0.45×10^6	43,000	95,000	95,000	21×10^6	95,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
2. From Orbit	5,700 to 6,000	1.0	- -	6 : 1	1	200	2/3	3 Days	740	1500	1500	0.21×10^6	45,000	72,000	72,000	11×10^6	72,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
D. Lunar Take-off																							
1. To Orbit	6,000 to 6,500	1.0 to 1.6	0.15	None	1	140	1/3	Weeks	1500	3000	3000	0.42×10^6	21,200	40,000	40,000	5.6×10^6	40,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
2. Direct to Earth	9,000 to 11,000	1.0 to 1.5	0.30	None	None	130	1	Weeks	930	3000	4500	0.59×10^6	13,600	40,000	60,000	7.8×10^6	60,000	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
IV. Mars Flights																							
A. Trajectory Correction																							
1. Midcourse - Flt.	50 to 1,000	0.025 to 0.50	0.05	None	6	300	1/3	250 Days	9470	10,000	500	0.15×10^6	23,750	25,000	1,250	0.30×10^6	1,250	A	E	F	2 - A	100 - A	A
2. Terminal - Flt.	100 to 1,000	0.020 to 1.0	0.3	None	3	62	1/3	250 Days	9880	10,000	500	31,000	24,700	25,000	1,250	78,000	1,250	A	E	F	2 - A	100 - A	A
3. Midcourse - Return	50 to 1,000	0.025 to 0.50	0.05	None	6	310	1/3	3 Years	2840	3,000	150	45,600	8,000	8,500	425	0.13×10^6	425	A	E	F	2 - A	100 - A	A
4. Terminal - Return	200 to 1,500	0.03 to 1.0	0.1 to 0.25	None	3	62	1/3	3 Years	2960	3,000	150	9,300	8,380	8,500	425	26,400	425	A	E	F	2 - A	100 - A	A
B. Mars Orbit																							
1. No Atmospheric Braking	5,000 to 20,000	1.0 to 3.0	0.5 to 2.0	None	None	104	1	250 Days	3200	9000	18,000	1.6×10^6	8,500	23,500	47,000	4.2×10^6	47,000	M / V	M - B / V - B	M - F / V - F	1 - M / 4 - V	80 - M / 20 - V	V
2. With Atmospheric Deceleration	~ 1,000	0.5 to 1.0	0.03	None	5 to 10	30	1/3	250 Days	7900	9000	9,000	0.28×10^6	20,600	23,500	23,500	0.73×10^6	23,500	M	M - B	M - F	1 - M	100 - M	A
C. Mars Landing																							
1. Direct	13,000 to 21,000	1.0 to 2.0	- -	10 : 1	0 to 1	300	2/3	250 Days	1450	9000	18,000	2.2×10^6	3,800	23,500	47,000	5.75×10^6	47,000	M / V	M - B / V - B	M - F / V - F	1 - M / 4 - V	80 - M / 20 - V	V
2. From Orbit	11,000 to 15,000	2.0 to 4.0	- -	10 : 1	1	140	2/3	250 Days	630	3500	14,000	0.79×10^6	1,500	8,500	34,000	1.9×10^6	34,000	M / V	M - B / V - B	M - F / V - F	1 - M / 4 - V	80 - M / 20 - V	V
D. Mars Take-off																							
1. To Orbit	15,000 to 17,000	0.7 to 1.0	0.30	None	1	167	1/3	Years	1600	10,000	15,000	2.4×10^6	4,000	25,000	37,500	6.2×10^6	37,500	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
2. Direct to Earth	20,000 to 35,000	- -	0.30	None	None	- -	1	Years	- -	- -	- -	- -	- -	- -	- -	- -	- -	M / V	M - E / V - E	M - F / V - G	1 - M / 4 - V	80 - M / 20 - V	V
a. 1st Stage	- -	1.5 to 2.0	- -	- -	- -	100	- -	- -	8300	25,000	50,000	5×10^6	- -	- -	- -	- -	50,000	- -	- -	- -	- -	- -	- -
b. 2nd Stage	- -	1.0 to 2.0	- -	- -	- -	120	- -	- -	1600	8,300	16,600	2×10^6	- -	- -	- -	- -	16,600	- -	- -	- -	- -	- -	- -

NOTES FOR EACH COLUMN:

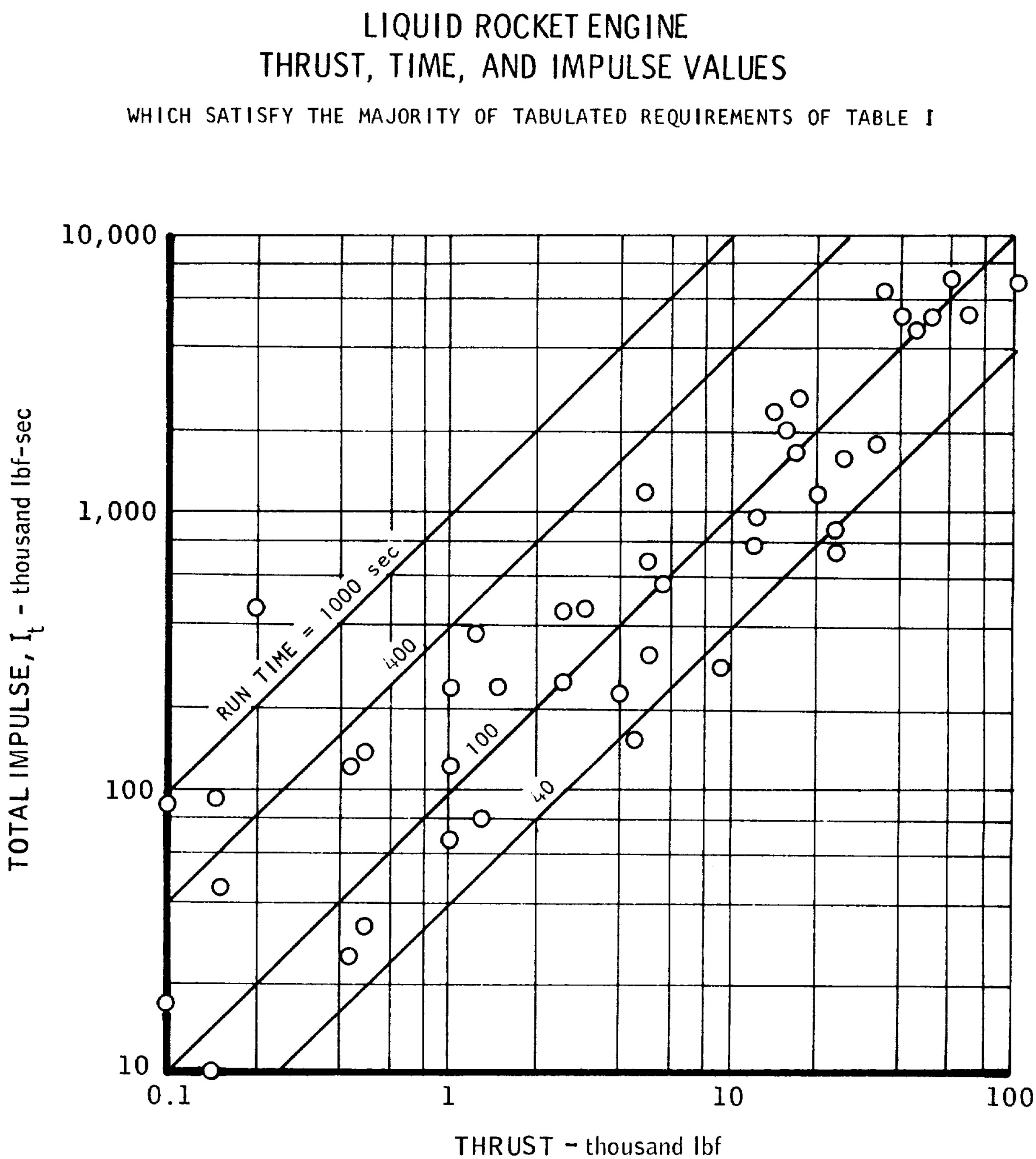
1. Estimated range of ideal velocity increments for each maneuver (Reference A-1).
2. The thrust and acceleration requirements for the various operations (Reference A-1).
3. The allowable total impulse error per unit mass at cutoff (Reference A-1).
4. Control of thrust level or total impulse per unit time (Reference A-1).
5. Possible number of restarts required (Reference A-1).
6. Typical burning time required for a fixed thrust level and total impulse maneuver (Reference A-1).
7. Typical thrust program for mission (Reference A-1). 1 = Constant, 2 = Variable, 3 = Pulse.
8. Propellant space storability requirements (Reference A-1).

9 - 12. Representative system requirements for a small payload (usually unmanned) using the Atlas-Centaur or Saturn launch vehicle (Reference A-1).
13 - 16. Representative system requirements for a large payload (potentially manned) using the Saturn or Nova launch vehicle (Reference A-1).
17. Maximum thrust requirements for a payload and given mission launched with the Nova H-8 vehicle.
18. Type engine used for mission / maneuver. M = Main, V = Vernier, A = Attitude.
19. Possible configuration installation of engines. B = Buried, E = Exposed.
20. Possible configuration attachment of engines. F = Fixed, G = Gimballed.
21. Typical cluster arrangement.
22. Distribution of thrust levels between engines to accomplish thrust variability and control.
23. Mode of thrust correction and control in boost phase. V = Vernier, A = Attitude control.

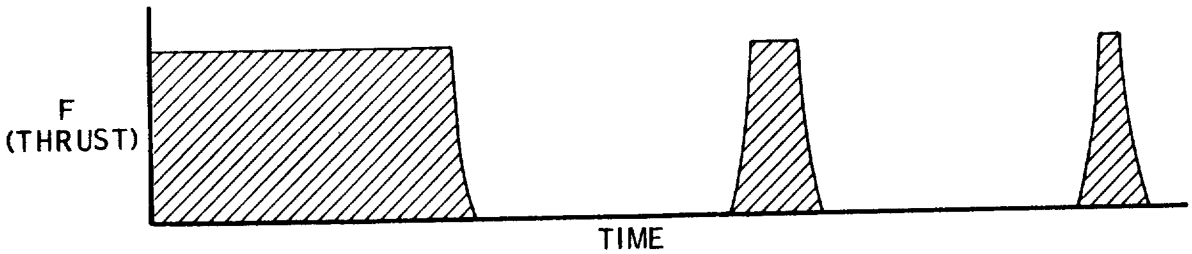
TABLE II

EFFECT OF PROPELLANT CHOICE ON COOLING TECHNIQUE APPLICABILITY

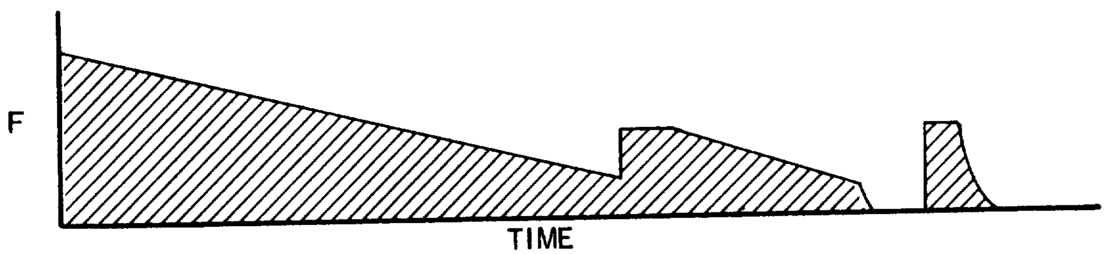
Propellants	Maximum I_{sp} at P_c = 300 psia, ξ = 40		Flame Temperature	Principal Exhaust Products	Regenerative Convection Cooling Capacity of Fuel (Typical)	Applicable Cooling and Material Concepts for Each Propellant Grouping
	I_{sp}	O/F				
Earth Storable, Hypergolic					(Nucleate Boiling)	
N_2O_4/N_2H_4	342	1.4	5700°R	N_2, H_2, H_2O	$(\frac{q}{A})_{lim}$ = 12 Btu/in.^{2} sec at 600 psia, 30 fps	● Regenerative cooling with Aerozine-50 and (N_2H_4 + 10% EDA)-Reference: JPL-TR32-109 ● Radiation cooling, coated refractory metals at 3300°F.
N_2O_4/0.5 N_2H_4- 0.5 UDMH (AEROZINE-50)	338	2.0	5700°R	N_2, H_2, H_2O, CO	$(\frac{q}{A})_{lim}$ = 6 Btu/in.^{2} sec at 600 psia, 30 fps	● Heat sink for times to 2 min or longer (Coated graphite, pyrolytic graphite, silicon carbide). ● Ablative (Silica-phenolic) ● Film and transpiration at high chamber pressures.
Cryogenic						
O_2/H_2	456	4.5	5600°R	H_2, H_2O	Convective cooling with H_2	● Regenerative cooling with H_2. ● Open tube cooling with H_2 (Above 10,000 pound thrust).
F_2/H_2	478	10	7300°R	HF, H_2, H	$\frac{q}{A}$ = 15 Btu/in.^{2} sec at 600 psia	● Radiation cooling - O_2/H_2 to 3300°F with coated refractories - F_2/H_2 to above 4000°F, uncoated refractories - OF_2/H_2 limited to low chamber pressures (less than 50 psi)
OF_2/H_2	477	7.0	6600°R	HF, H_2, H_2O, H	V = 1580 fps ΔT = 3000°F	● Heat sink - for times to 2 min, limited chamber pressure (pyrolytic and isotropic graphite, carbides) ● Film and transpiration - H_2 good coolant, reacts with graphite at high temperature.
Space Storable						
OF_2/B_2H_6	437	4.0	8000°R	HF, H, BOF, H_2	Unsuitable	● Radiation cooling - limited to pressures below 20 psi or exit nozzle skirt at large values of A/A* ● Heat sink - limited run time, best materials unknown. Most promising material pyrolytic graphite.
OF_2/CH_4	420	5.3	7600°R	HF, CO, H	Unsuitable	● Ablative - limited run time (Throat insert required) ● Film and transpiration - coolant capabilities of propellants unknown.


LIQUID ROCKET ENGINE
THRUST, TIME, AND IMPULSE VALUES
WHICH SATISFY THE MAJORITY OF TABULATED REQUIREMENTS OF TABLE I
10,000
1,000
100
10
0.1
1
10
100
RUN TIME = 1000 sec
400
100
40
TOTAL IMPULSE, I_t - thousand lbf-sec
THRUST - thousand lbf

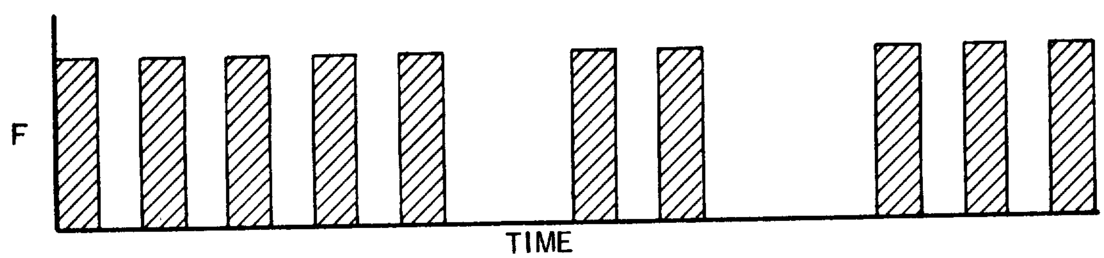
TYPICAL THRUST TIME PLOTS FOR SPACE ENGINE MISSIONS



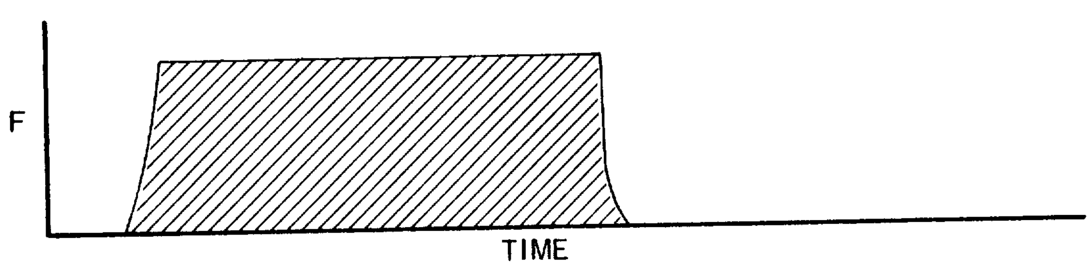
CONSTANT THRUST, VARIABLE IMPULSE
(LUNAR LANDING)



VARIABLE THRUST, VARIABLE IMPULSE
(RENDEZVOUS)



PULSE ROCKET -- DISCRETE IMPLUSE BITS
(ATTITUDE CONTROL)



CONSTANT THRUST, ONE START IN SPACE ENVIRONMENT
(LUNAR TAKE-OFF)

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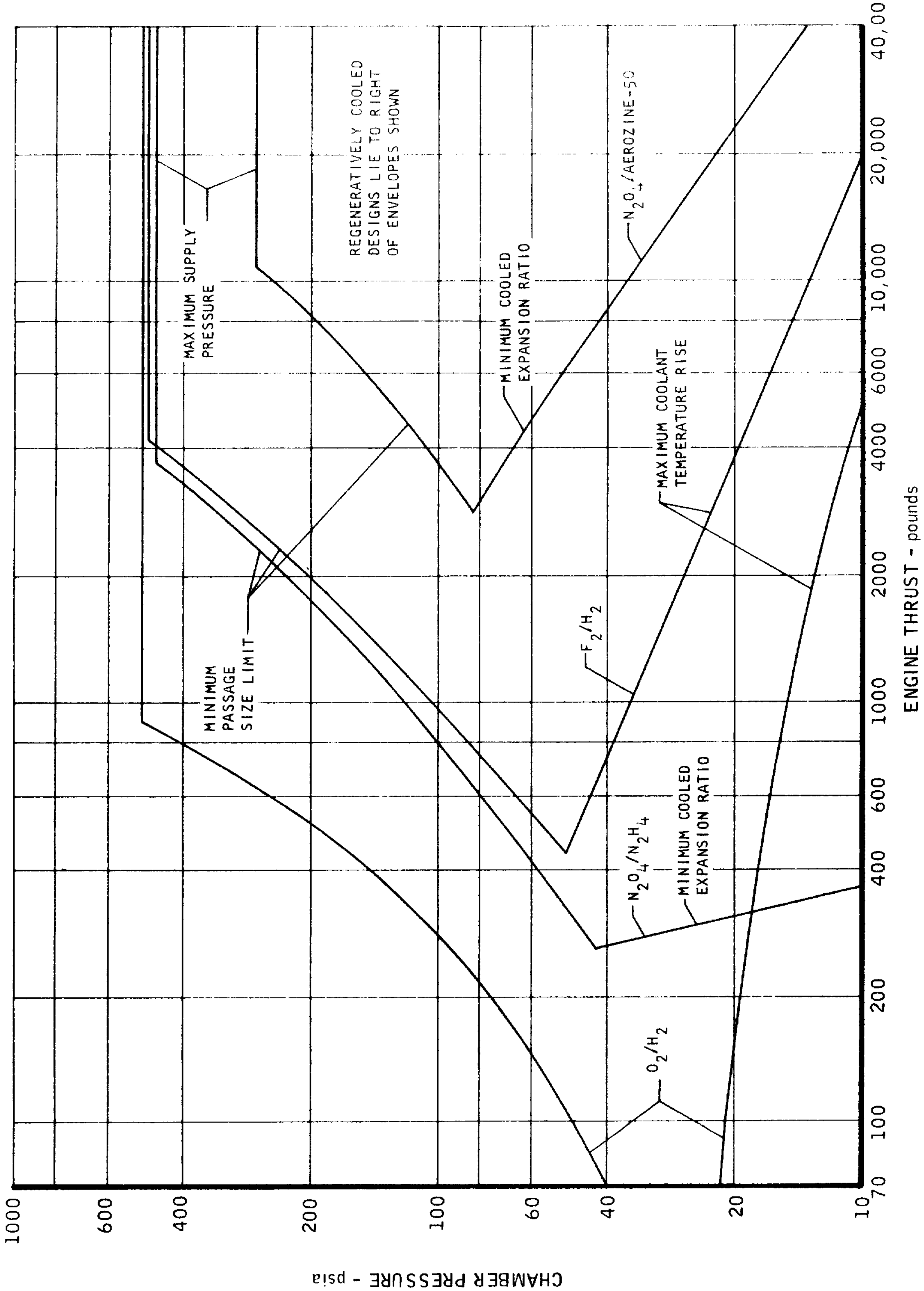
COOLING METHOD SCREENING CHART

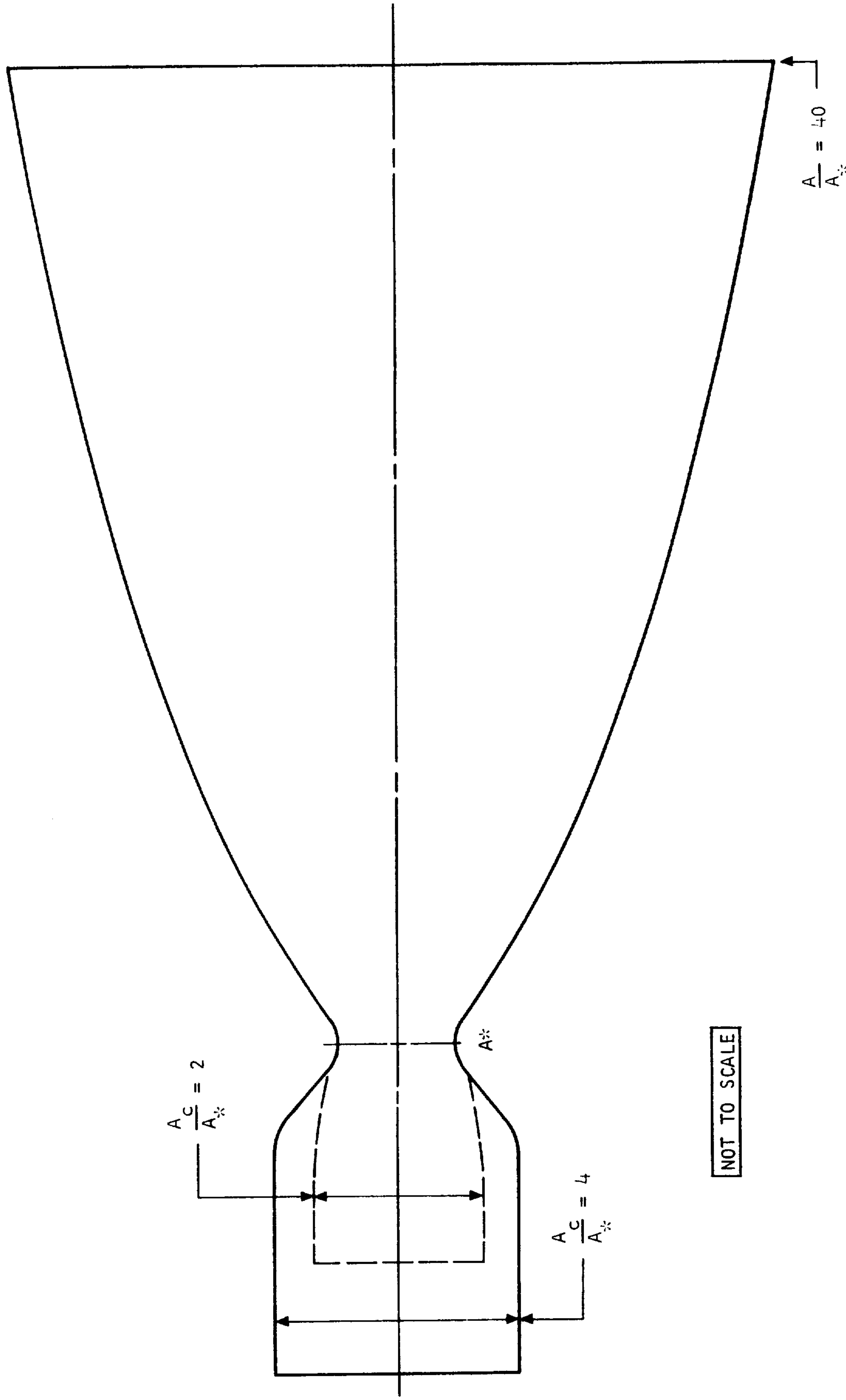
COOLING METHOD	APPLICABLE TO SPACE RESTART OF MOTOR	APPLICABLE TO SHORT PULSE MOTOR	APPLICABLE TO THROTTLEABLE MOTOR	RUN TIME LIMIT	EFFECT ON PROPELLANT CHOICE	CHAMBER PRESSURE LIMITS	VACUUM OPERATION	ATTITUDE WITH RESPECT TO SUN	METEOROIDS	HEAT TRANSFER TO VEHICLE	ADVANCED NOZZLE GEOMETRY	EXTERIOR ENGINE TEMPERATURE	PROPULSION SYSTEM PENALTIES
REGENERATIVE	APPLICABLE	NOT APPLICABLE	LIMITED RANGE	NO LIMIT	H_2 N_2H_4 N_2H_4 + EDA 0.5 N_2H_4-0.5 UDMH	AFFECTED BY COOLANT PASSAGE DESIGN	RESIDUAL THRUST	TRAPPED COOLANT AFFECTED BY SOAK	MAY PUNCTURE TUBES	MINIMUM	LIMITED BY PASSAGE SIZES	APPROACHES COOLANT TEMPERATURE 500° to 1000°F	PRESSURE LOSSES
RADIATION	APPLICABLE	APPLICABLE	APPLICABLE	HOURS	COMPATIBILITY WITH WALLS AND COATING CRITICAL	50 psia OR LESS, TO 90 psia FOR LOW THRUST	POSSIBLE LIMIT ON COATING LIFE	NO EFFECT	MAY ERODE COATINGS	MAXIMUM	LIMITED BY PRESSURE & CONFIGURATION	3300°F MAXIMUM	LARGE CHAMBER SIZE DUE TO LOW P_c
ABLATION	APPLICABLE	APPLICABLE	APPLICABLE	5 to 20 MIN	LIMITS RUN TIME	LIMITS RUN TIME FOR THROAT APPLICATION	SOME RESIDUAL THRUST	SOAK TEMPERATURES LIMITED TO 500°F	NO EFFECT	LONG SOAK TRANSIENTS	THROAT EROSION CRITICAL	500° to 800°F MAXIMUM	CHAMBER WEIGHT AND RUN TIME LIMIT
FILM	LIMITED	NOT APPLICABLE	POSSIBLY LIMITED	NO LIMIT	COOLING PROPERTIES IMPORTANT	NO LIMIT	RESIDUAL THRUST	NO EFFECT	NO EFFECT	MINIMUM	APPLICABLE	CAN BE CONTROLLED	I_{sp} LOSS
TRANSPIRATION	LIMITED	NOT APPLICABLE	POSSIBLY LIMITED	LIMITED BY SOURCE OF COOLANT	COOLING PROPERTIES IMPORTANT	NO LIMIT	RESIDUAL THRUST	NO EFFECT	NO EFFECT	MINIMUM	APPLICABLE	CAN BE CONTROLLED	I_{sp} LOSS
OPEN TUBE	APPLICABLE	LIMITED	APPLICABLE	NO LIMIT	H_2 BEST	NO LIMIT	NO LIMIT	NO EFFECT	MAY PUNCTURE TUBES	COOLANT TEMPERATURE MAY BE > 1500°F	APPLICABLE LIMITED BY PASSAGE SIZE	MAY APPROACH 1500°F	AFFECTS OPTIMUM O/F
INERT HEAT SINK	APPLICABLE	APPLICABLE	APPLICABLE	LESS THAN 2 MINUTES	COMPATIBILITY WITH WALLS AND COATINGS CRITICAL	LIMITS RUN TIME	NO LIMIT	NO EFFECT	NO EFFECT	FUNCTION OF TIME AND SOAK TRANSIENTS	TIME LIMITED	CAN BE LIMITED. MAY BE > 4000°F	TIME LIMIT VS. CHAMBER WEIGHT
ENDOTHERMIC HEAT SINK	LIMITED	LIMITED	LIMITED	LIMITED BY SOURCE OF COOLANT	FLAME TEMPERATURE IMPORTANT	LIMITS RUN TIME	RESIDUAL THRUST	SOAK TEMPERATURES LIMITED	NO EFFECT	CAN BE LIMITED	TIME LIMITED	CAN BE LIMITED	TIME LIMITED

FIGURE 3

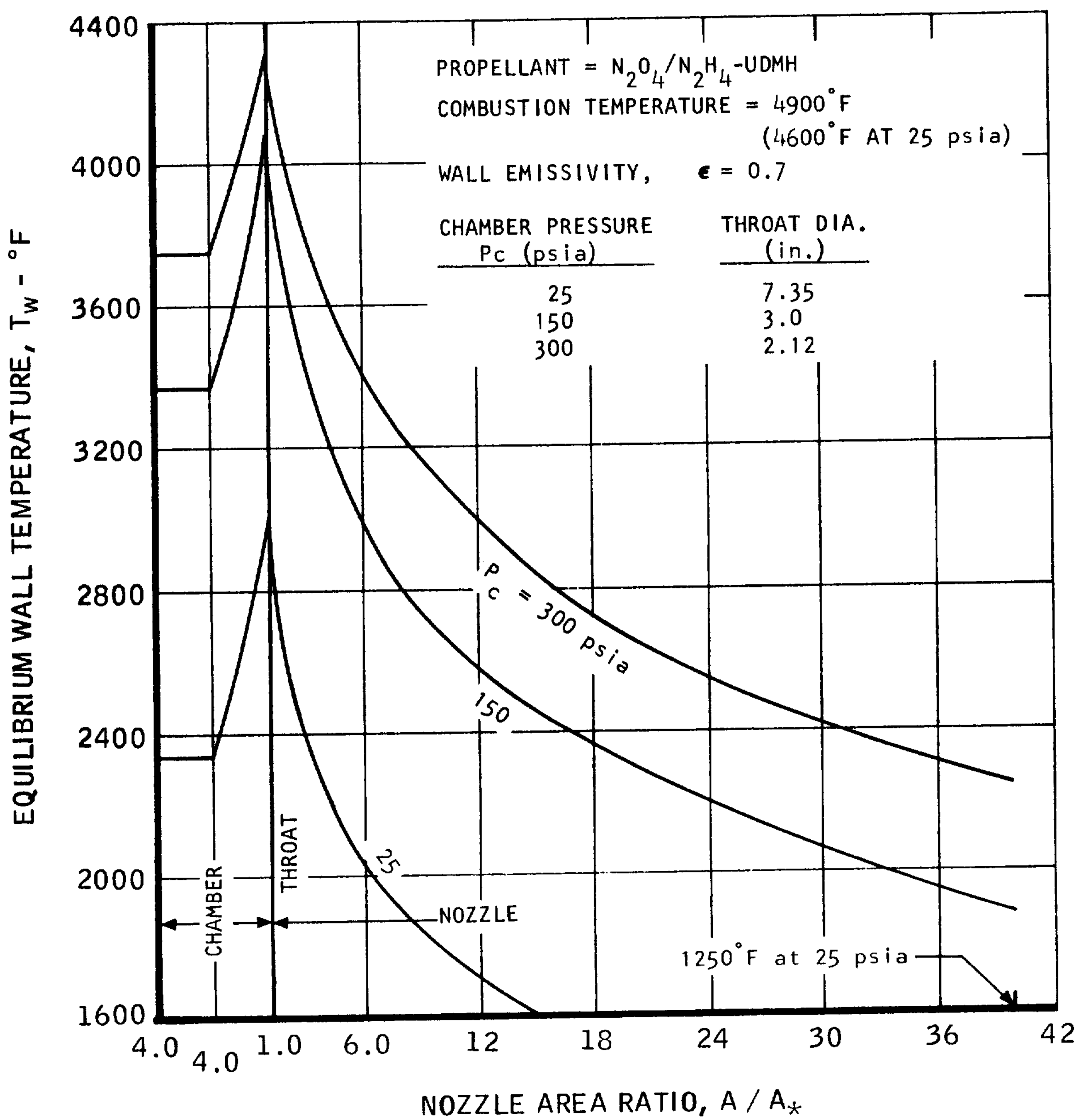
FEASIBILITY MAP FOR REGENERATIVE COOLING WITH
THE PROPELLANTS O_2 / H_2, F_2 / H_2, N_2O_4 / N_2H_4, AND N_2O_4 / AEROZINE-50

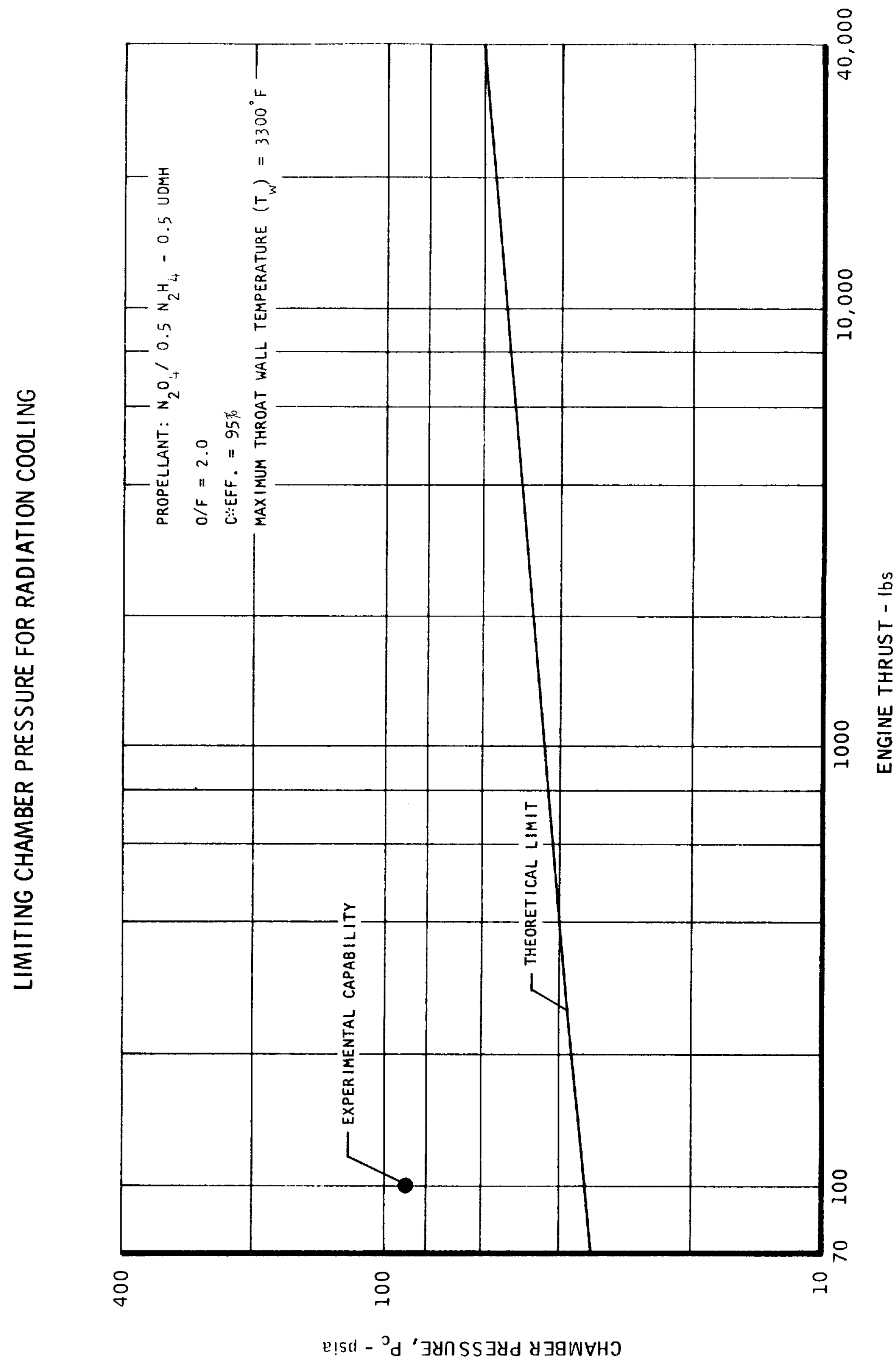


TYPICAL THRUST CHAMBER CONFIGURATIONS FOR SPACE ENGINE APPLICATION



EQUILIBRIUM WALL TEMPERATURES FOR THIN WALL RADIATION COOLED CHAMBER AND EXIT NOZZLE vs. NOZZLE AREA RATIO




LIMITING CHAMBER PRESSURE FOR RADIATION COOLING
PROPELLANT: N_2O_4 / 0.5 N_2H_4 - 0.5 UDMH
O/F = 2.0
C* EFF. = 95%
MAXIMUM THROAT WALL TEMPERATURE (T_w) = 3300°F
EXPERIMENTAL CAPABILITY
THEORETICAL LIMIT
400
100
10
CHAMBER PRESSURE, P_c - psia
70
100
1000
10,000
40,000
ENGINE THRUST - lbs

COMPILATION OF TEST DATA FOR ABLATIVE REFRASIL - PHENOLIC
CHAR DEPTH vs. BURNING TIME

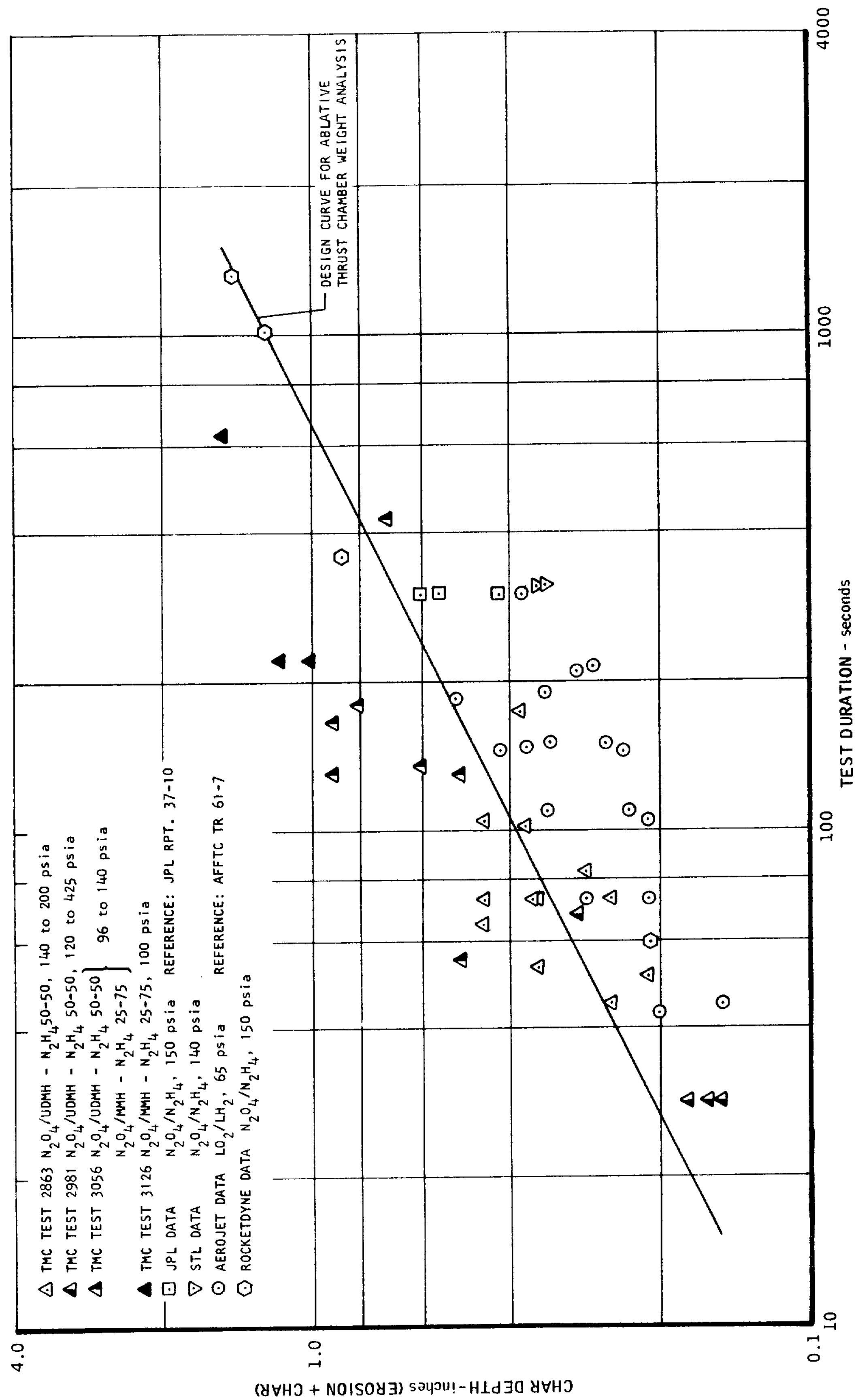


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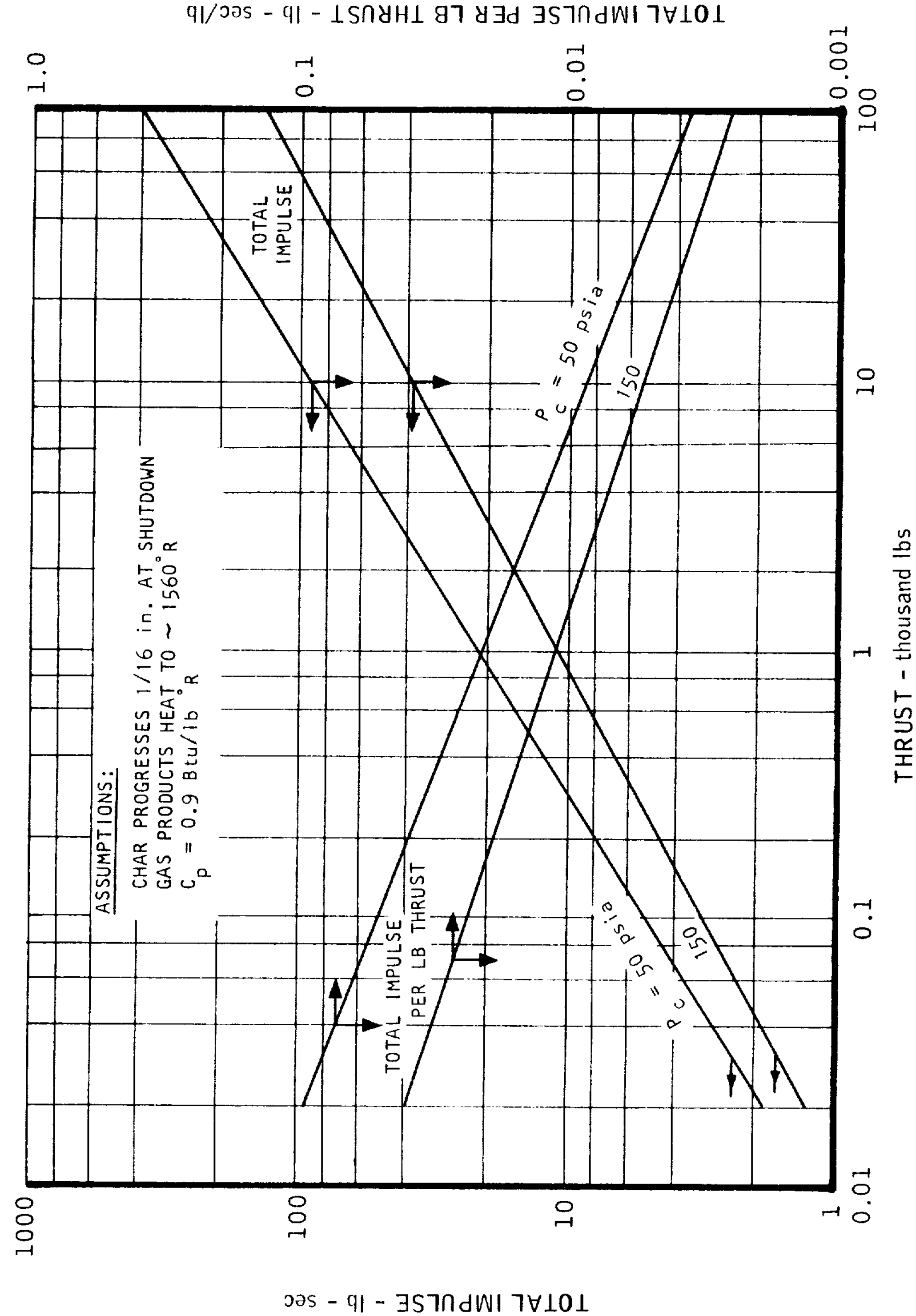
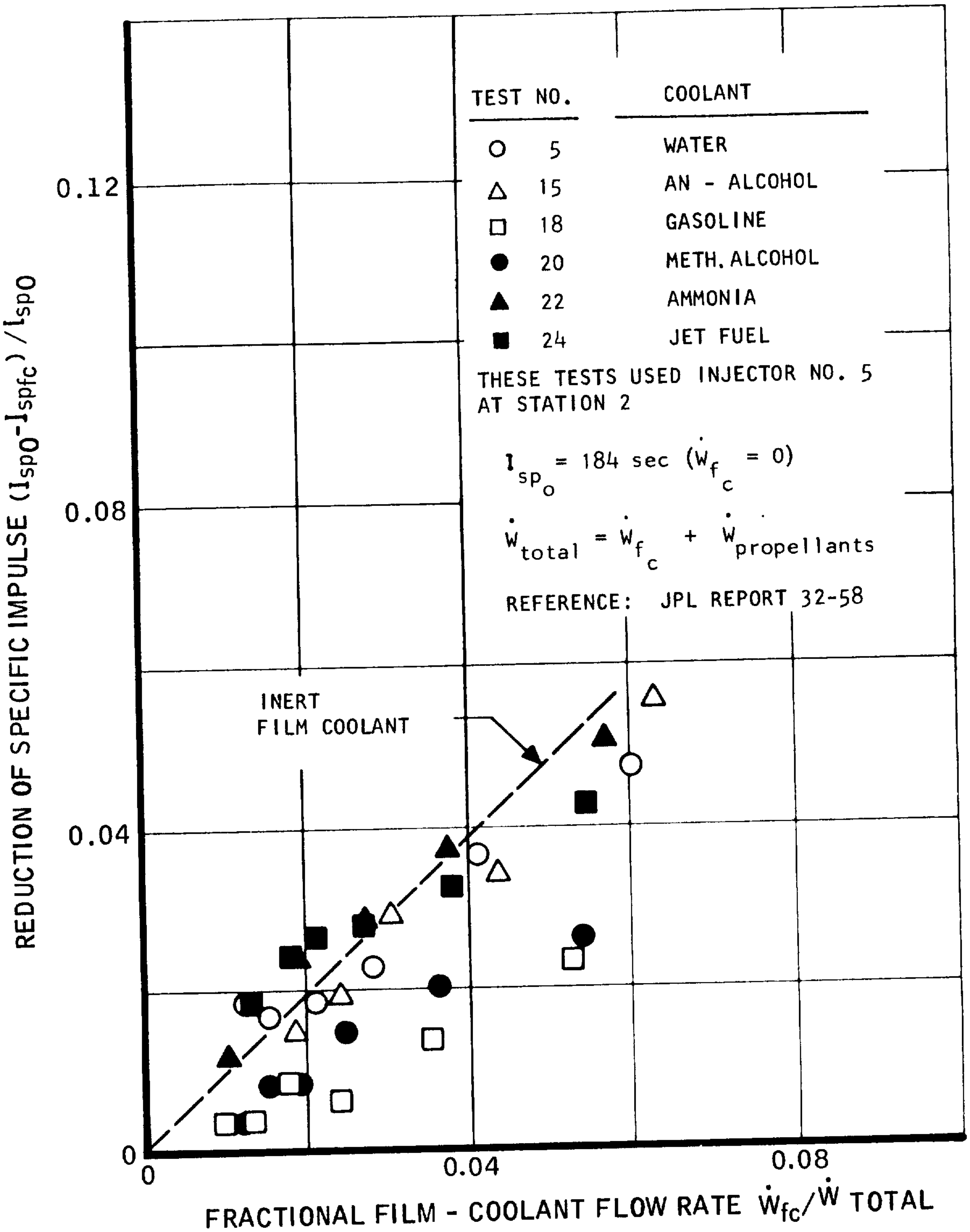
FIGURE 8

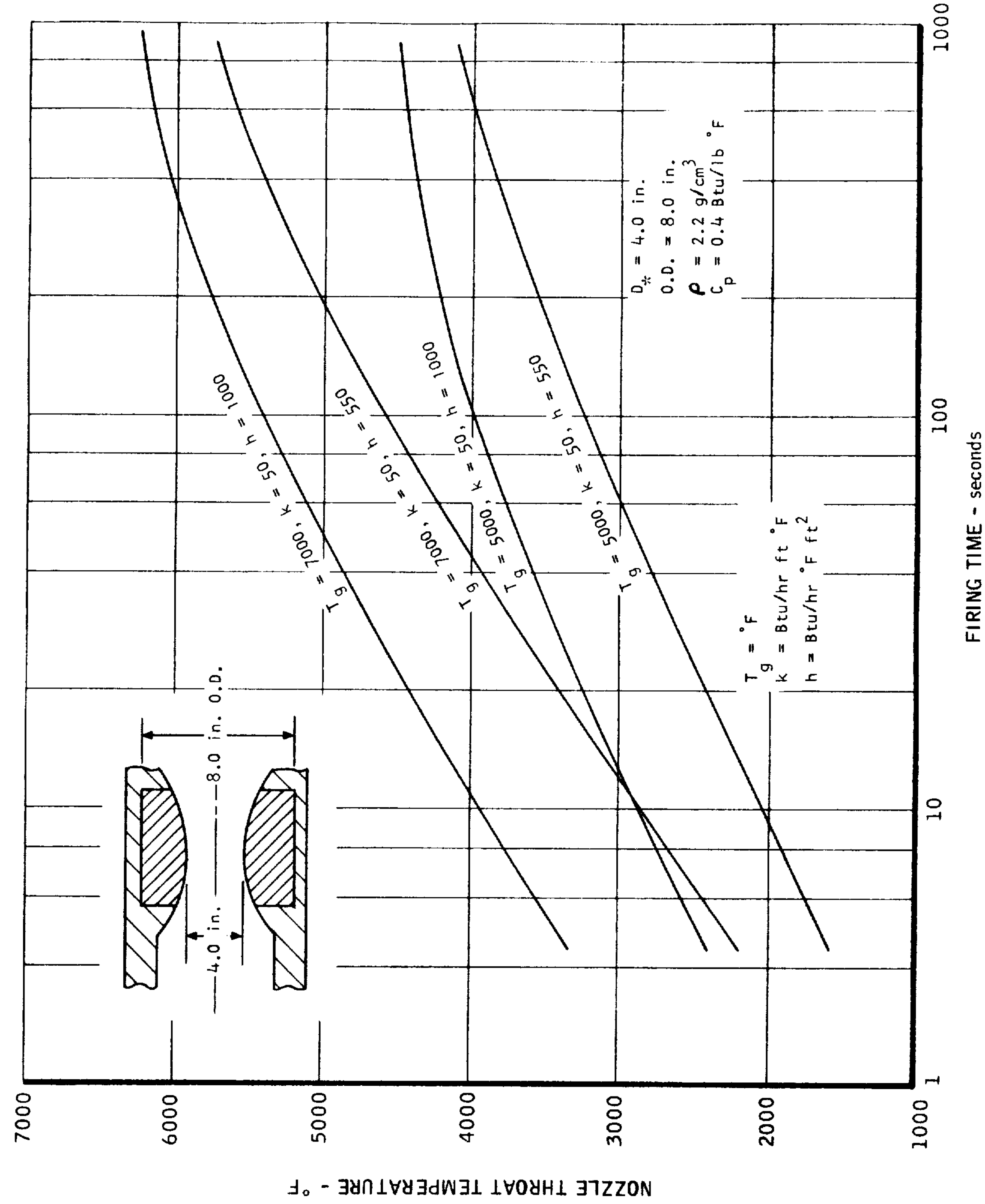
RESIDUAL TOTAL IMPULSE DUE TO POSTRUN CHARRING
OF A REINFORCED PHENOLIC THRUST CHAMBER vs. DESIGN THRUST



DECREASE OF MOTOR PERFORMANCE WITH FILM COOLING



TEMPERATURE RESPONSE OF UNCOOLED HEAT SINK EXIT NOZZLE INSERTS



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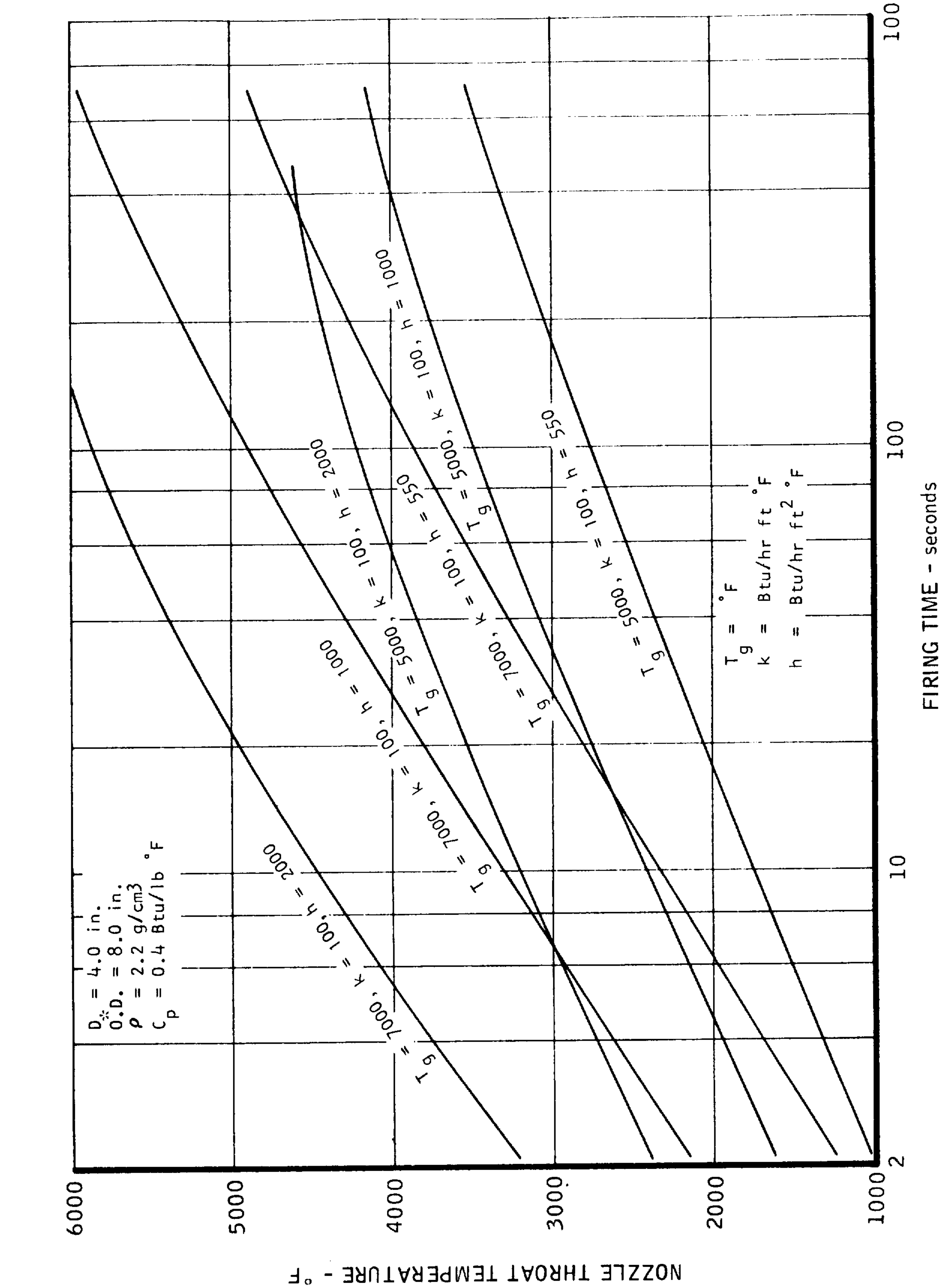
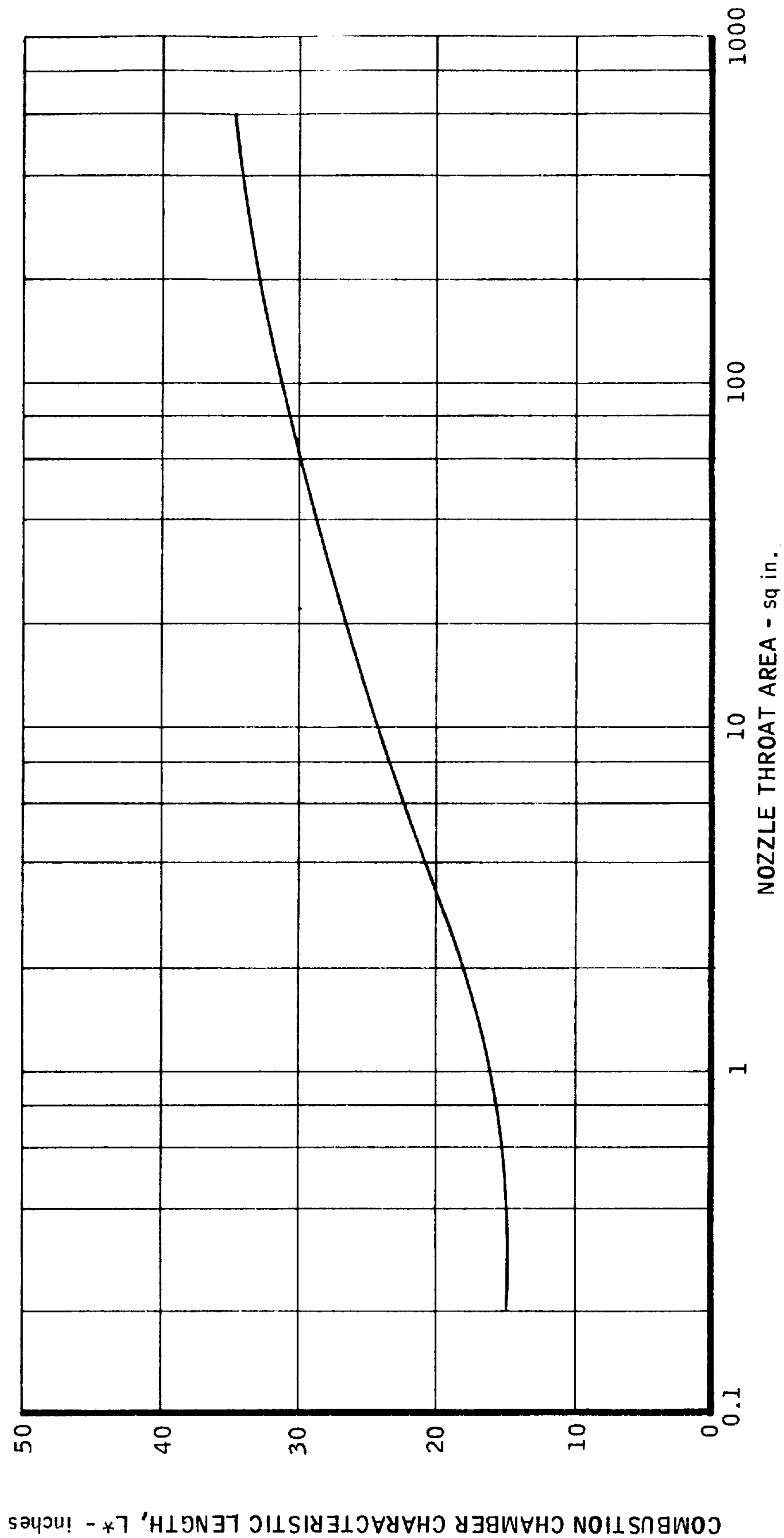
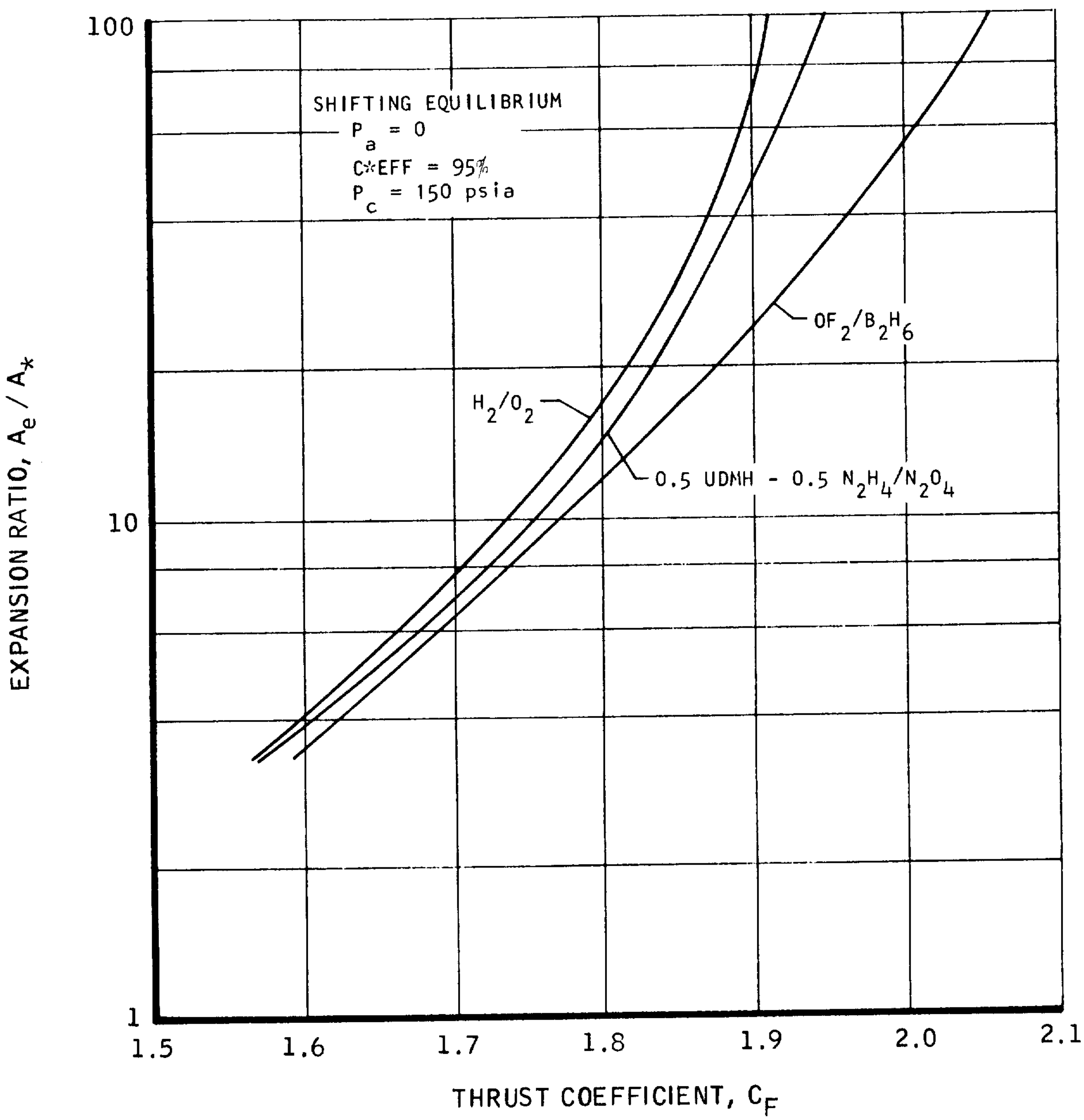
TEMPERATURE RESPONSE OF UNCOOLED HEAT SINK EXIT NOZZLE INSERTS



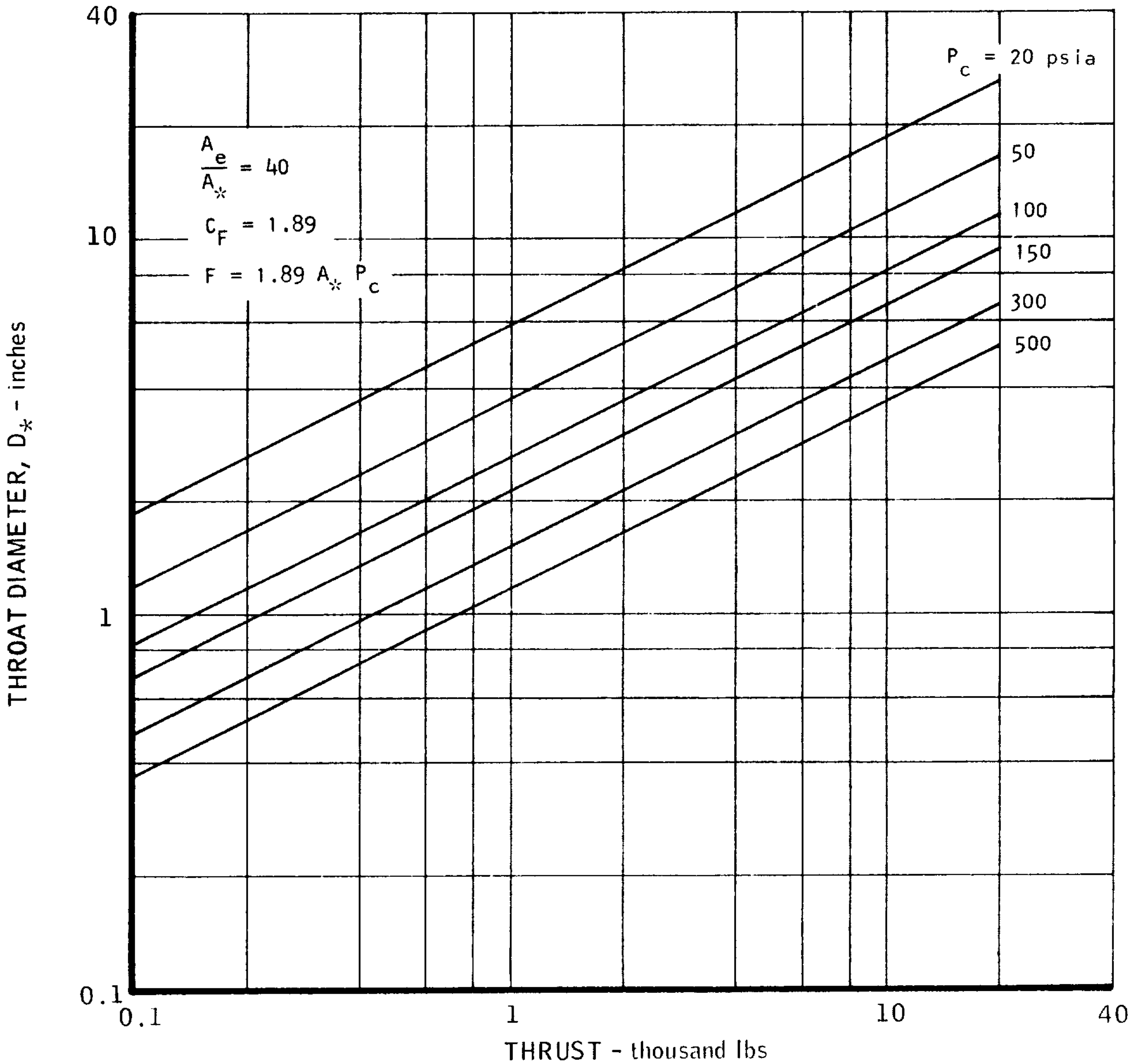
ASSUMED RELATIONSHIP BETWEEN L* AND THROAT AREA
BASED ON DATA FROM SEVERAL DEVELOPED THRUST CHAMBER DESIGNS



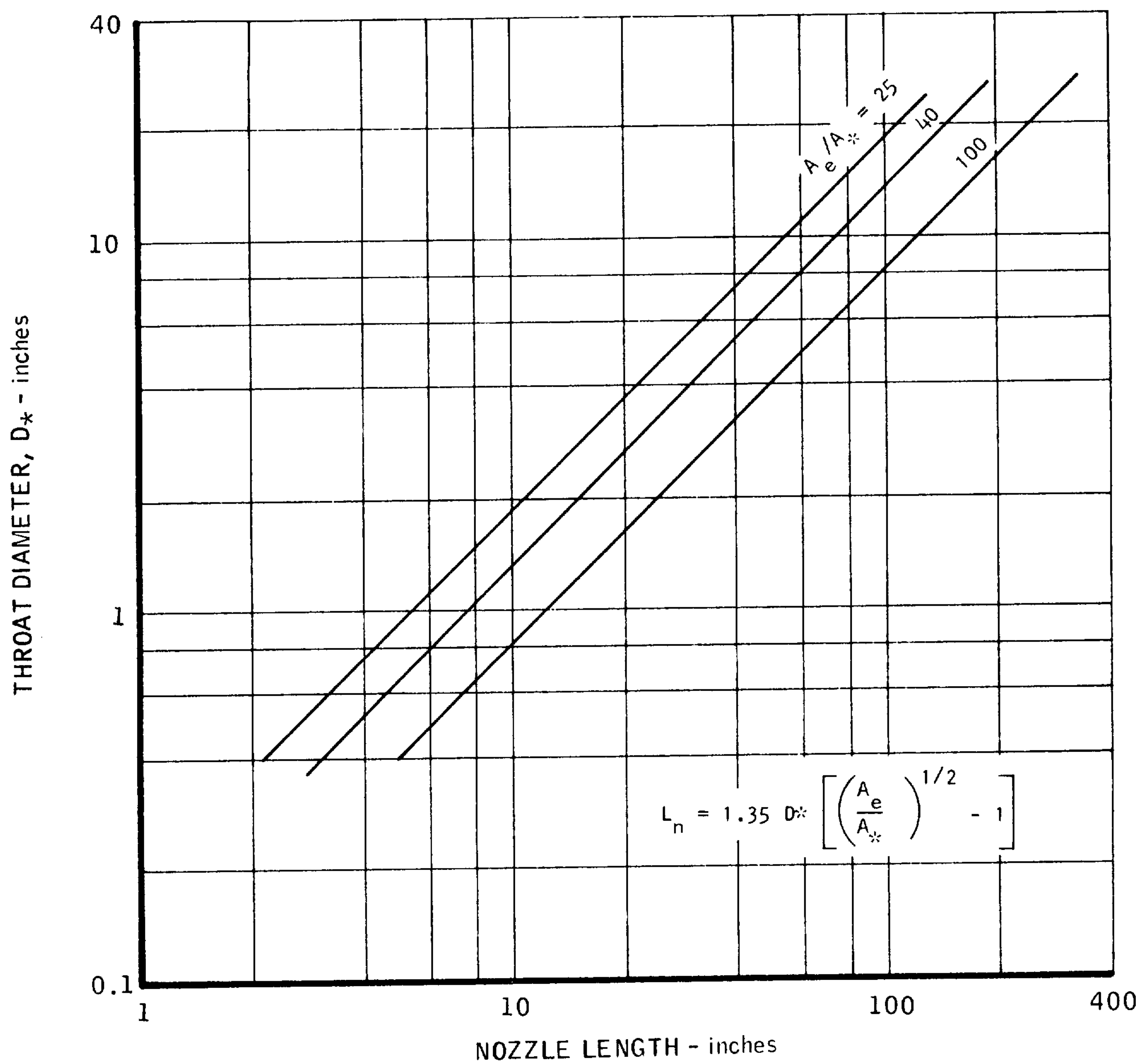
THRUST VARIATION WITH EXPANSION RATIO AND PROPELLANT



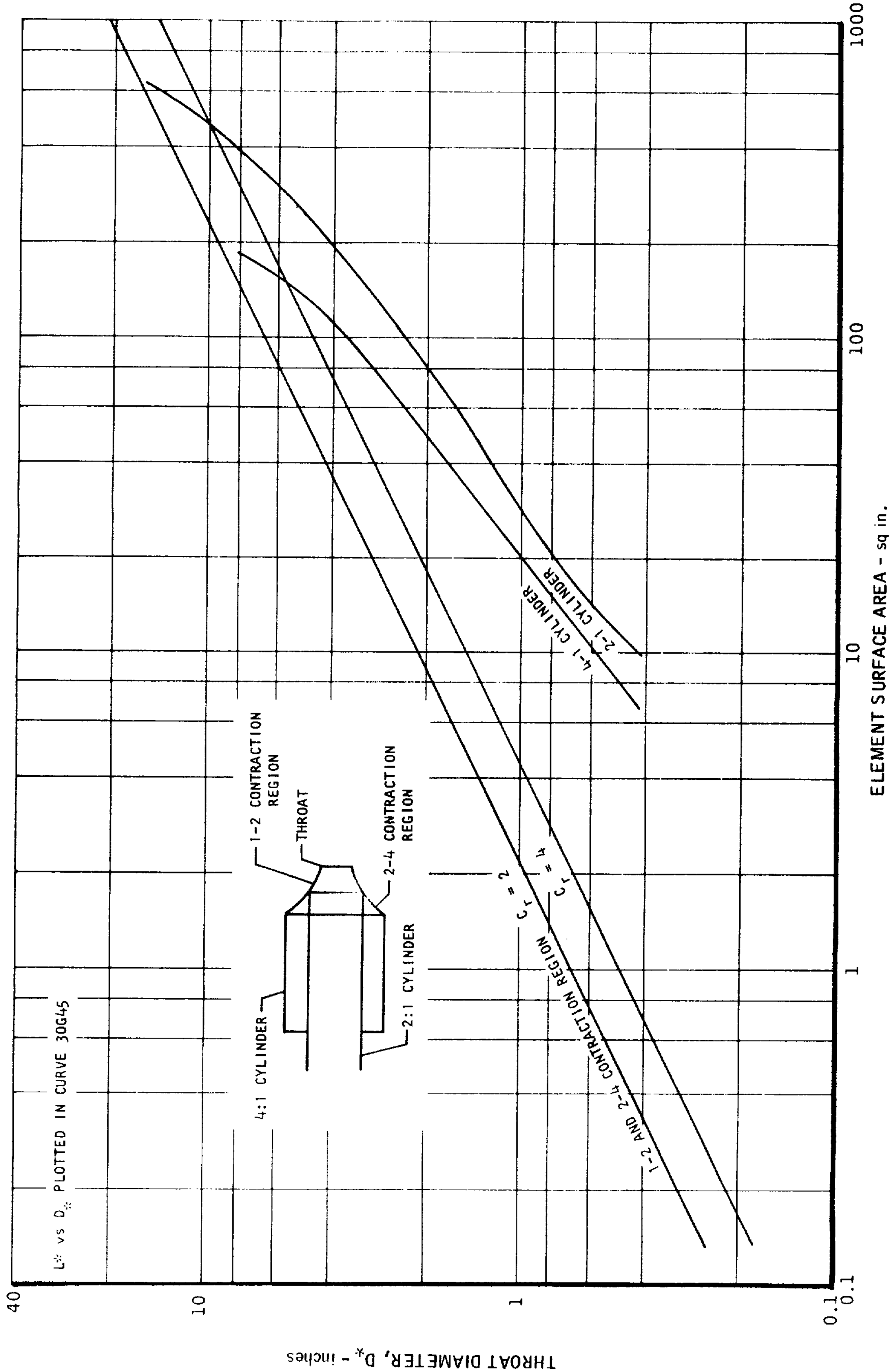
THRUST VARIATION WITH CHAMBER PRESSURE AND THROAT DIAMETER



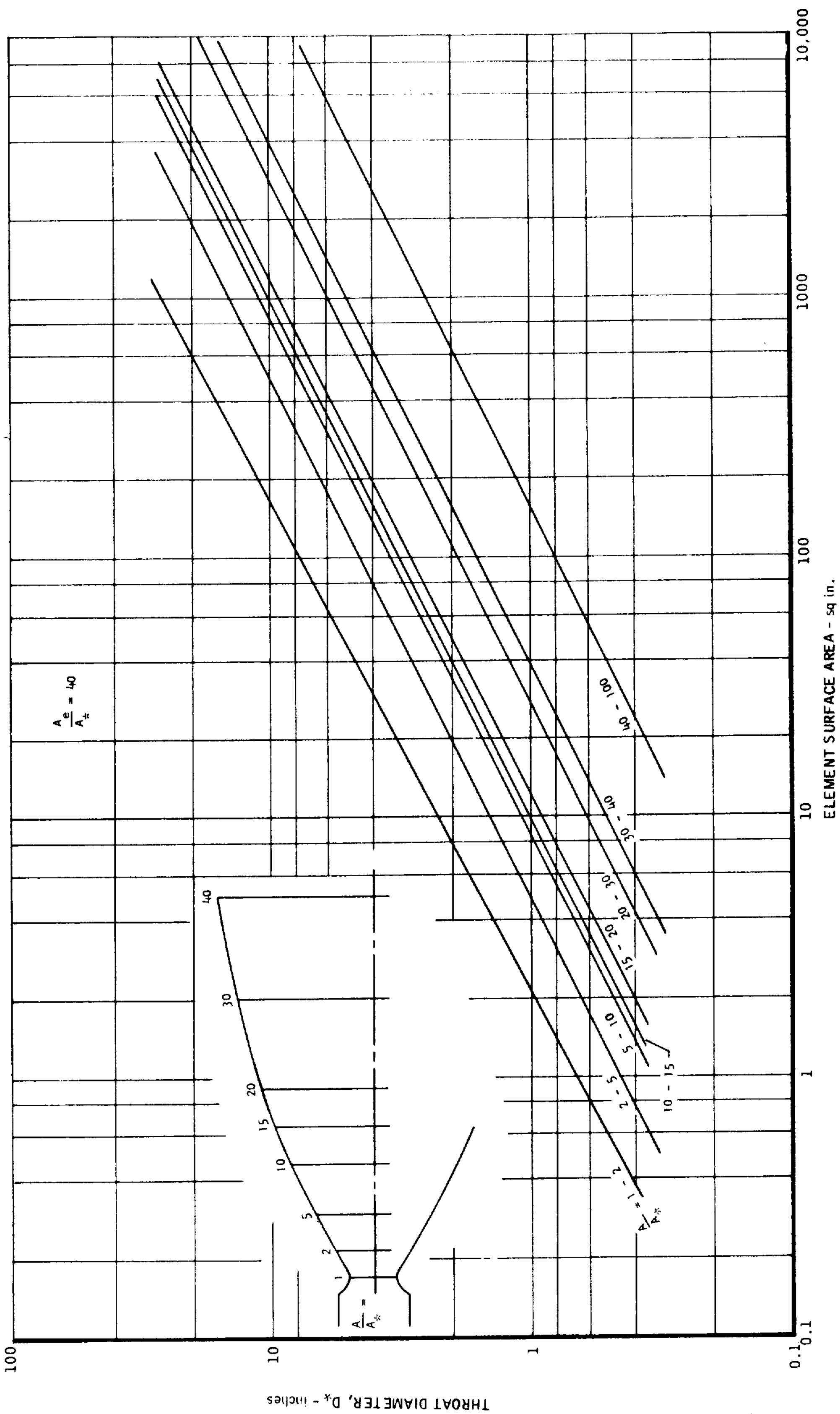
VARIATION OF EXPANSION NOZZLE LENGTH WITH THROAT DIAMETER



VARIATION OF SURFACE AREA OF COMBUSTION CHAMBER ELEMENTS WITH THROAT DIAMETER

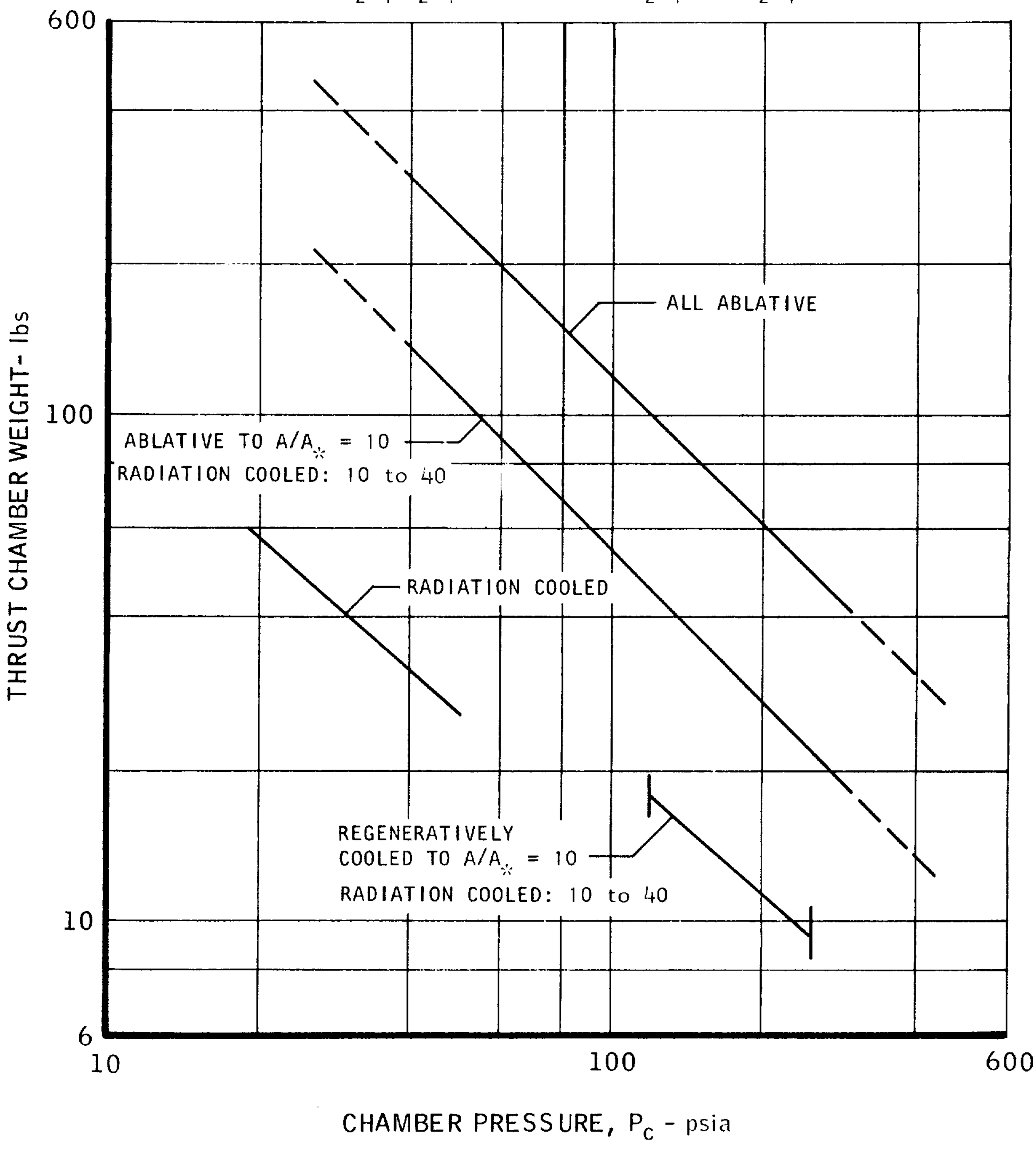


VARIATION OF SURFACE AREA OF EXPANSION NOZZLE ELEMENTS WITH THROAT DIAMETER

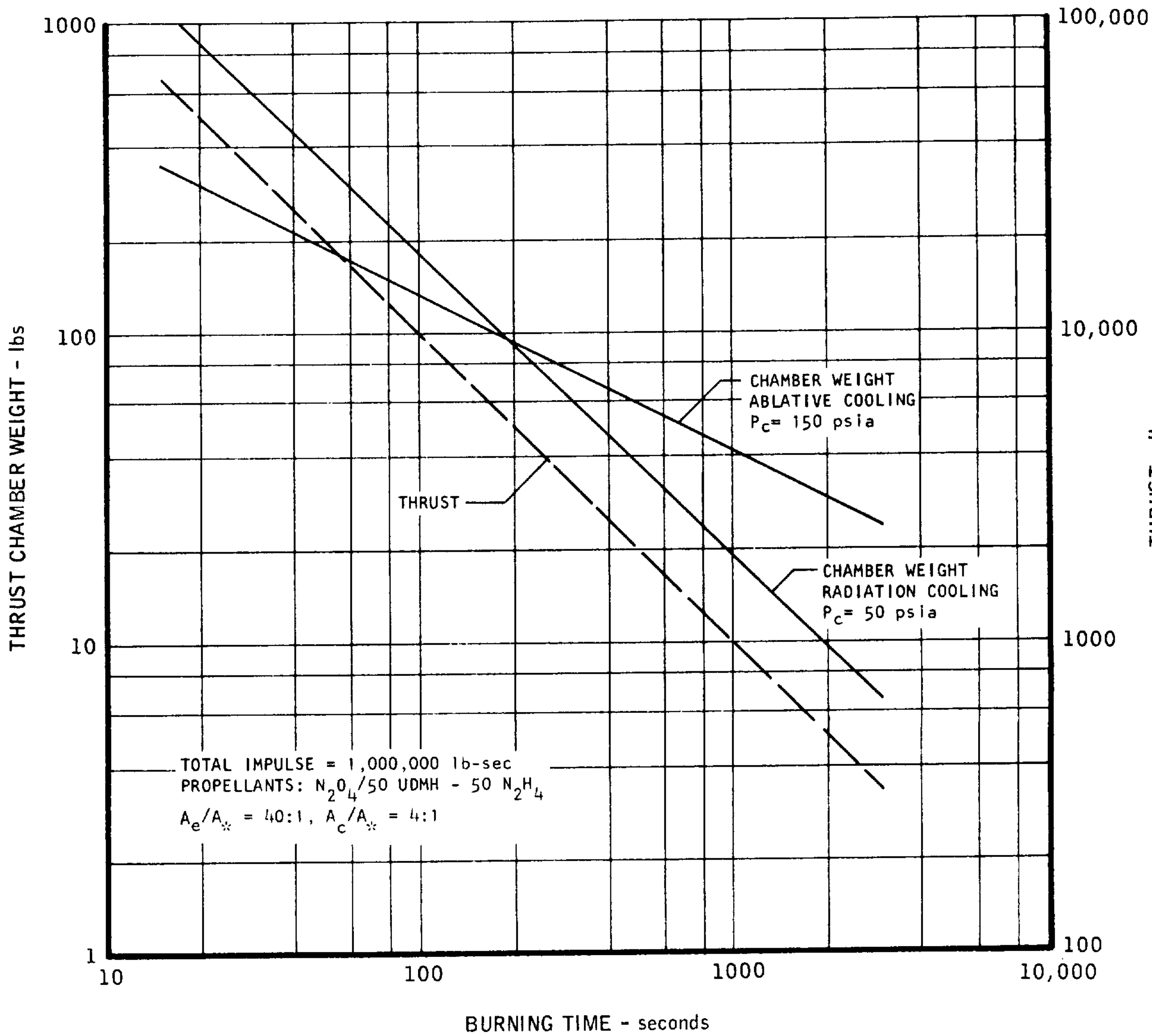


THRUST CHAMBER WEIGHTS FOR A LONG RUN, THROTTLING ENGINE

THRUST = 2000 to 500 lbs
$A_e/A_* = 40$
RUN TIME = 726 sec
PROPELLANTS = N_2O_4/N_2H_4 + 10% EDA or N_2O_4/0.5 N_2H_4 - 0.5 UDMH



THRUST CHAMBER WEIGHTS FOR CONSTANT TOTAL IMPULSE ENGINE

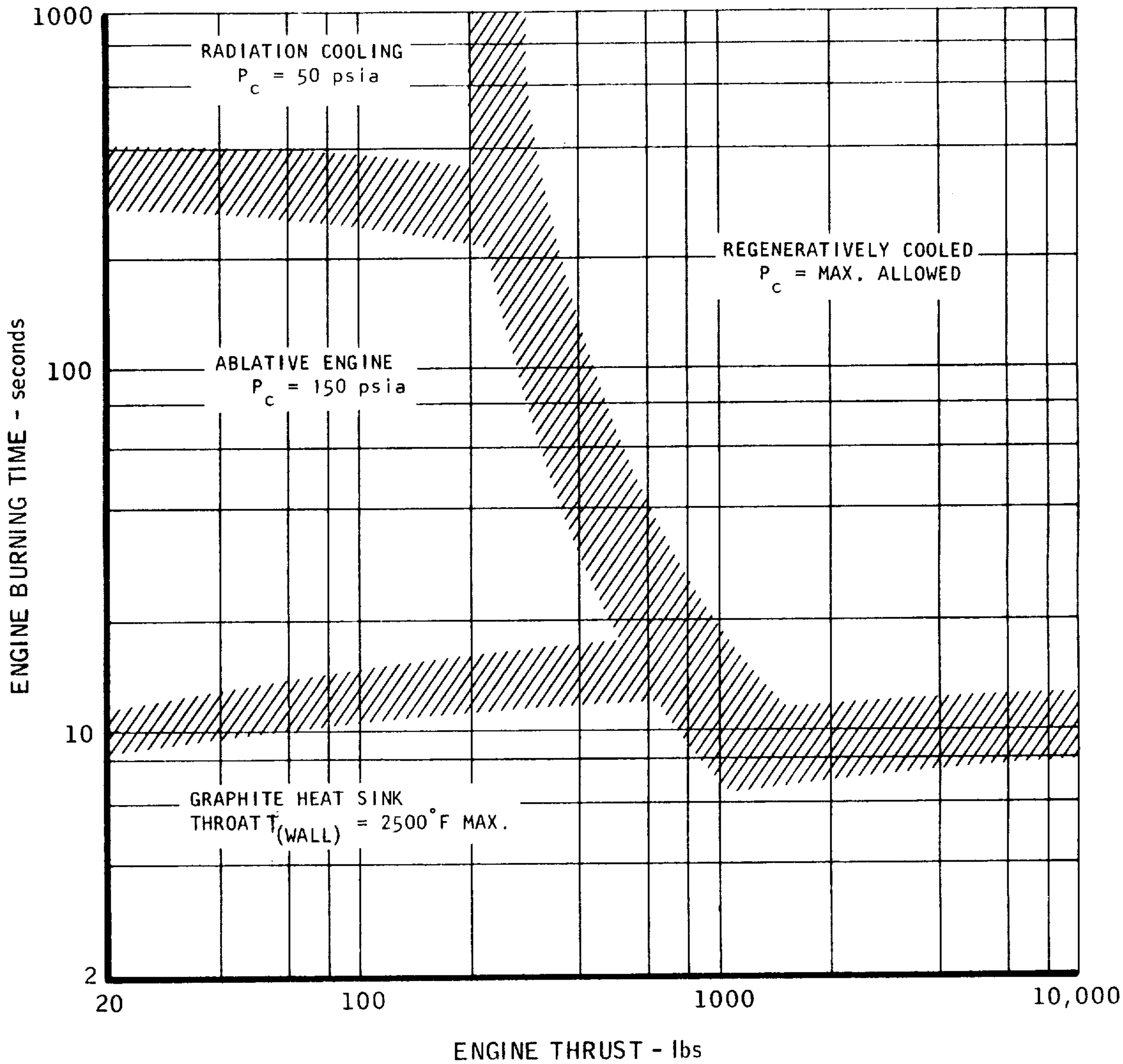


THRUST AND BURNING TIME ENVELOPES FOR MINIMUM WEIGHT SPACE ENGINES

PROPELLANT = $O_2 - H_2$

CONSTANT THRUST

$A_e/A_* = 40$



CONSTRUCTION USED IN WEIGHT ANALYSIS OF A
TYPICAL REGENERATIVELY COOLED THRUST CHAMBER

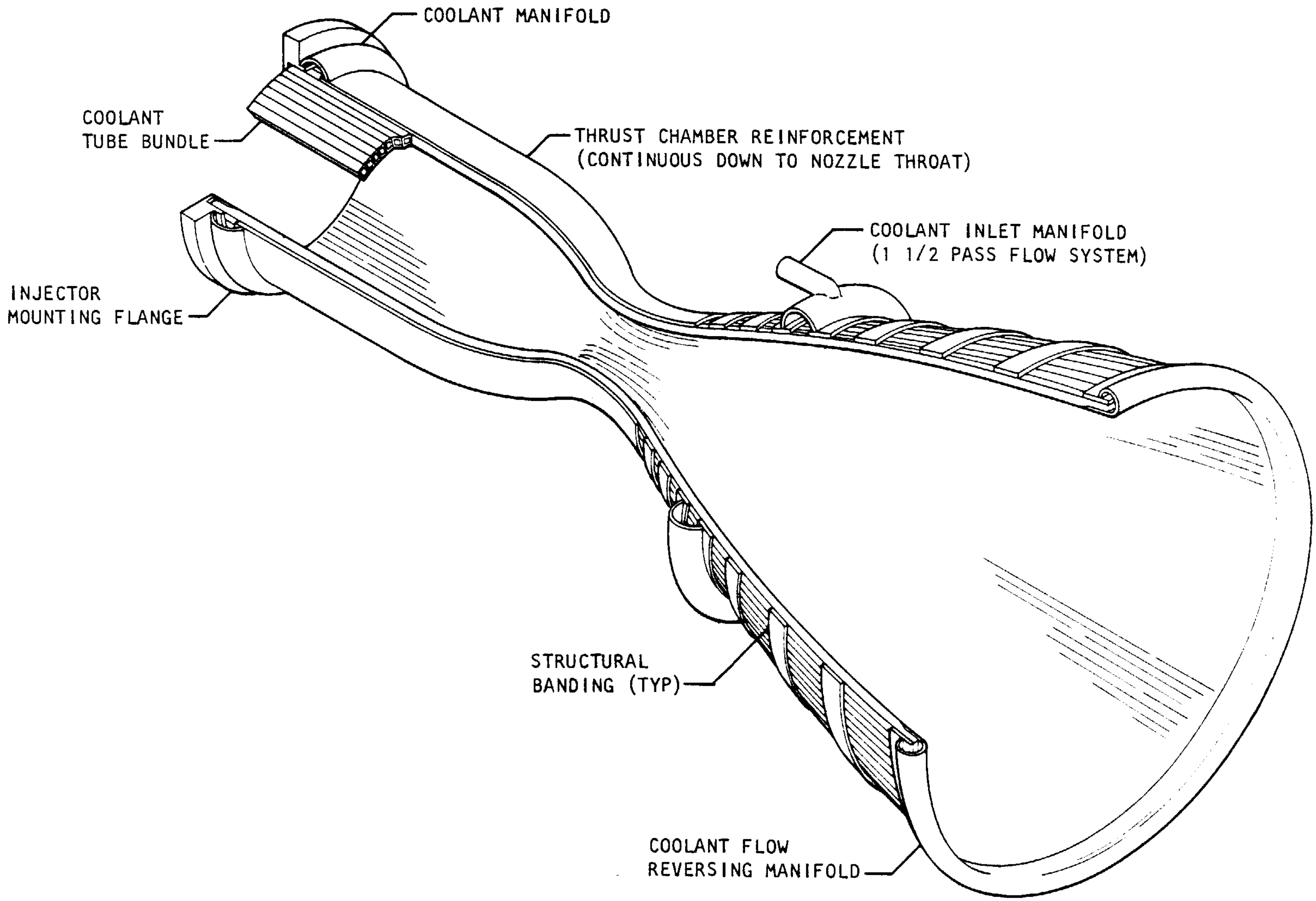
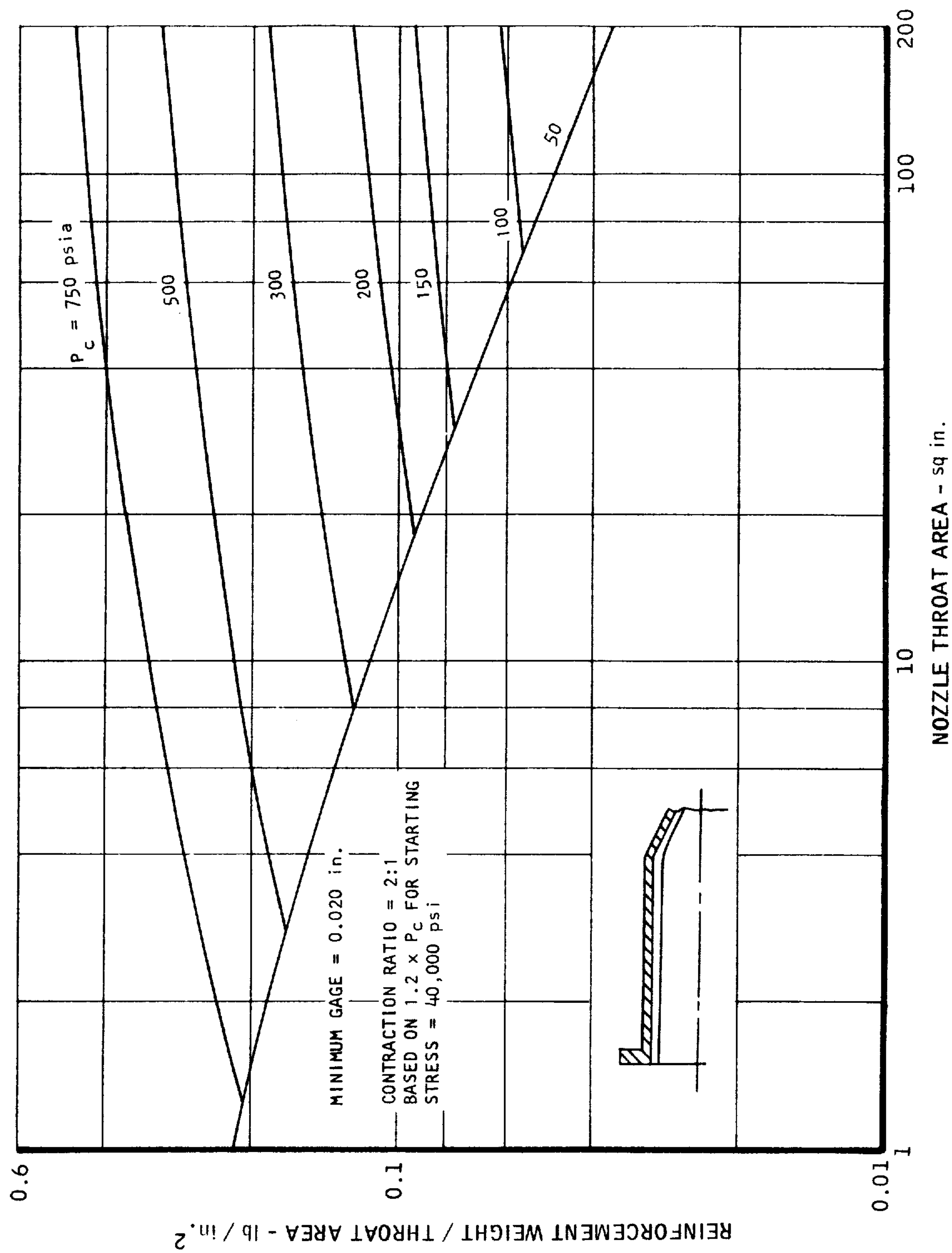


FIGURE 22

REINFORCEMENT WEIGHT UPSTREAM FROM NOZZLE THROAT
CONTRACTION RATIO 2:1



REINFORCEMENT WEIGHT UPSTREAM FROM NOZZLE THROAT
CONTRACTION RATIO 4:1

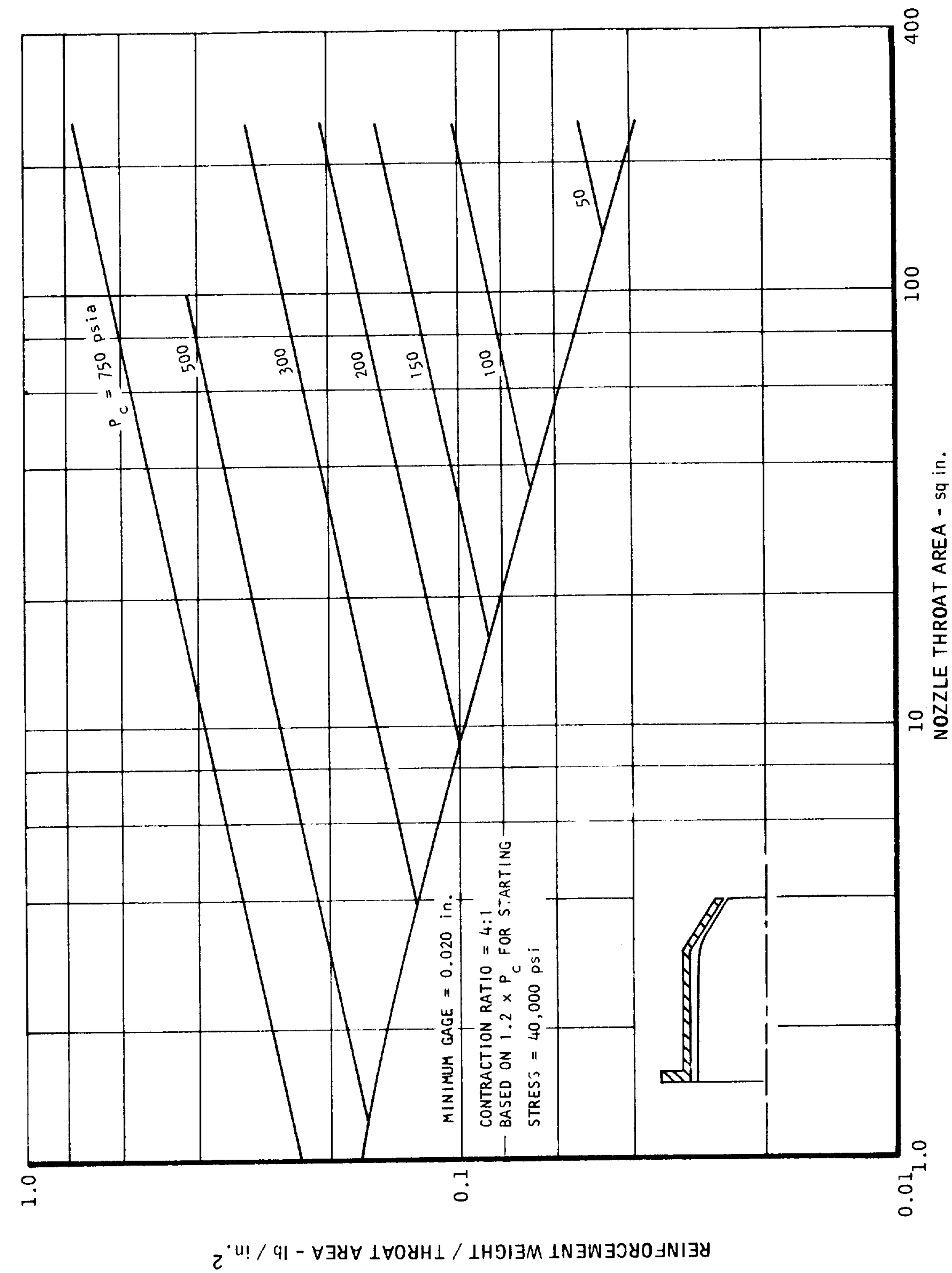
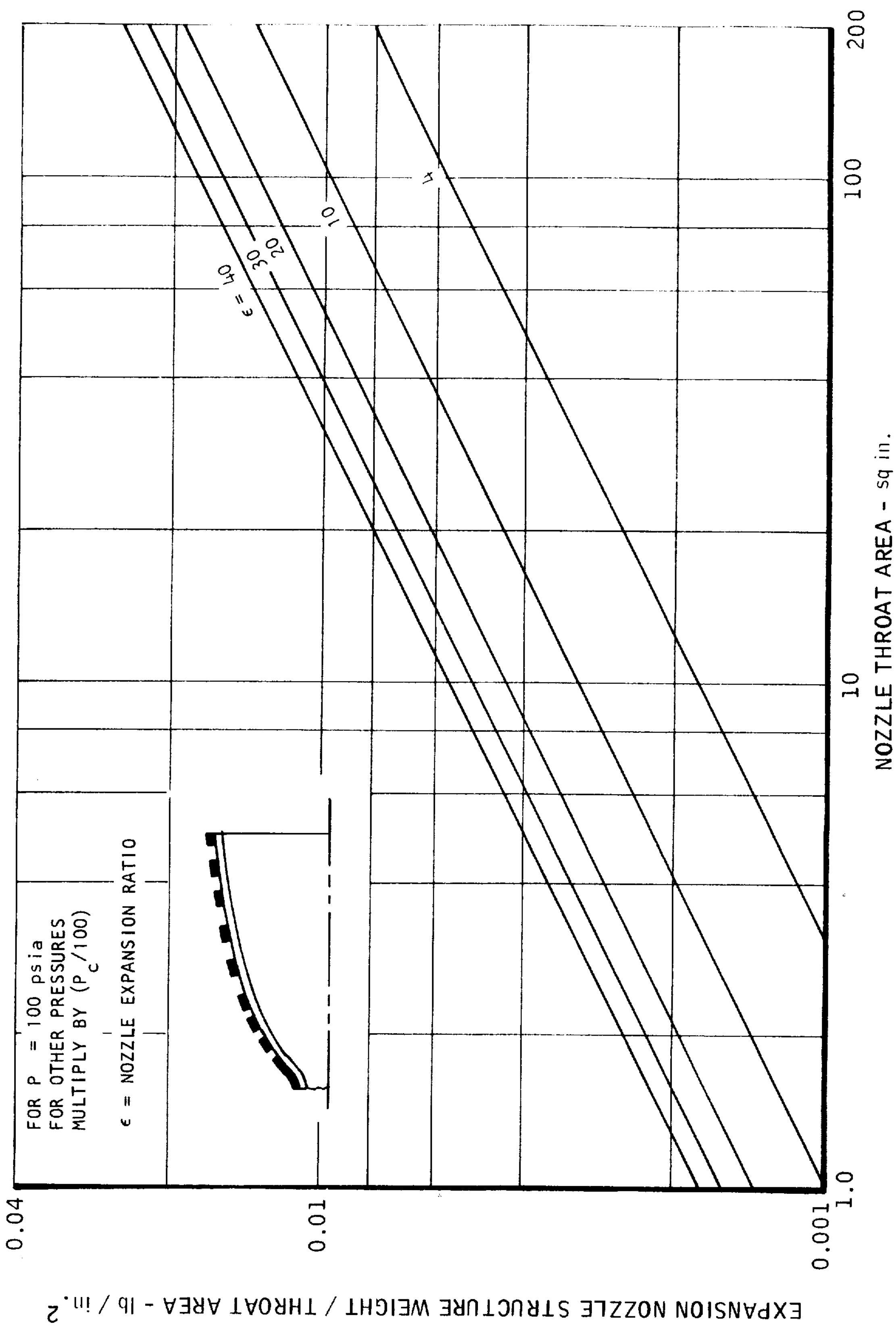
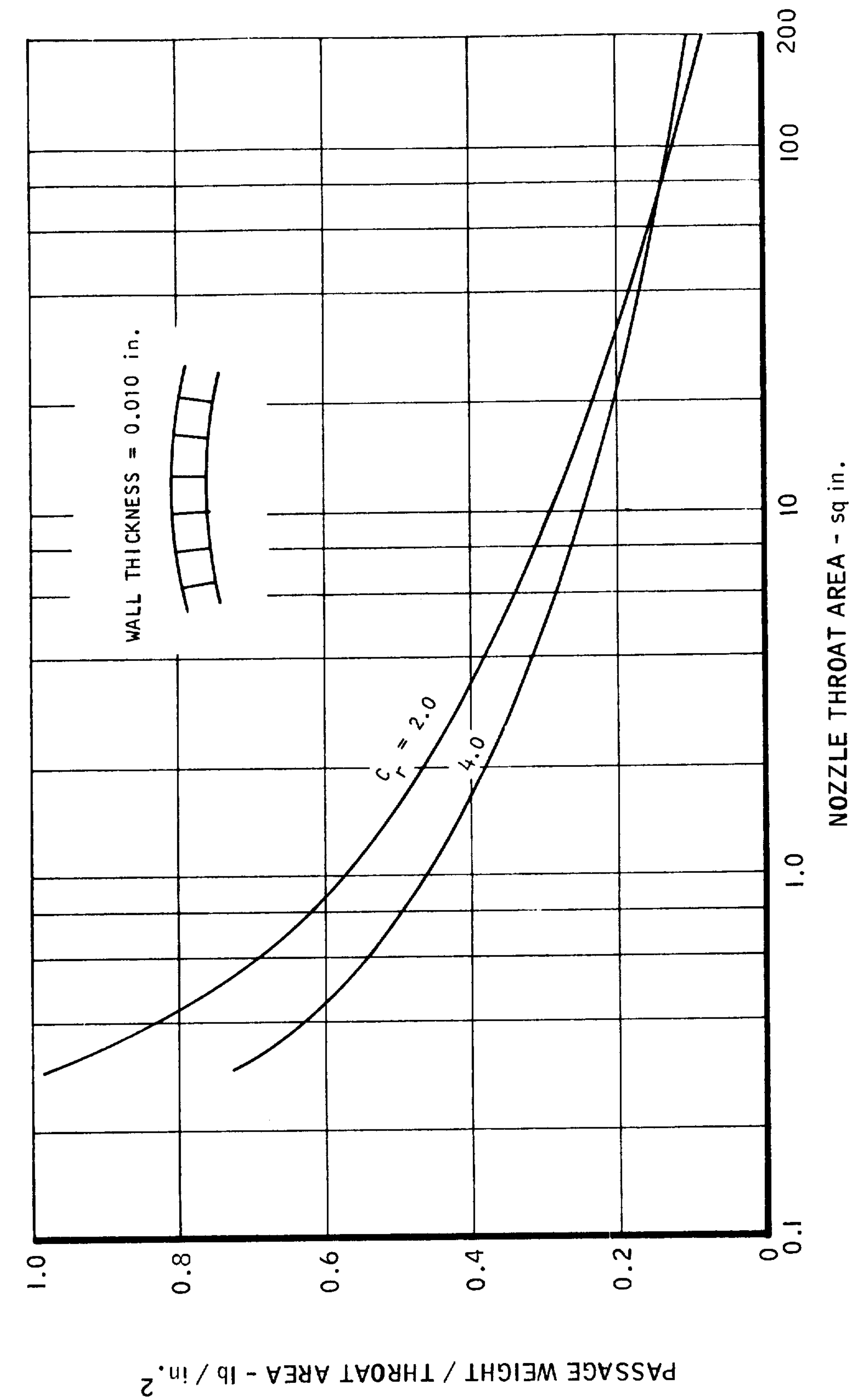


FIGURE 24

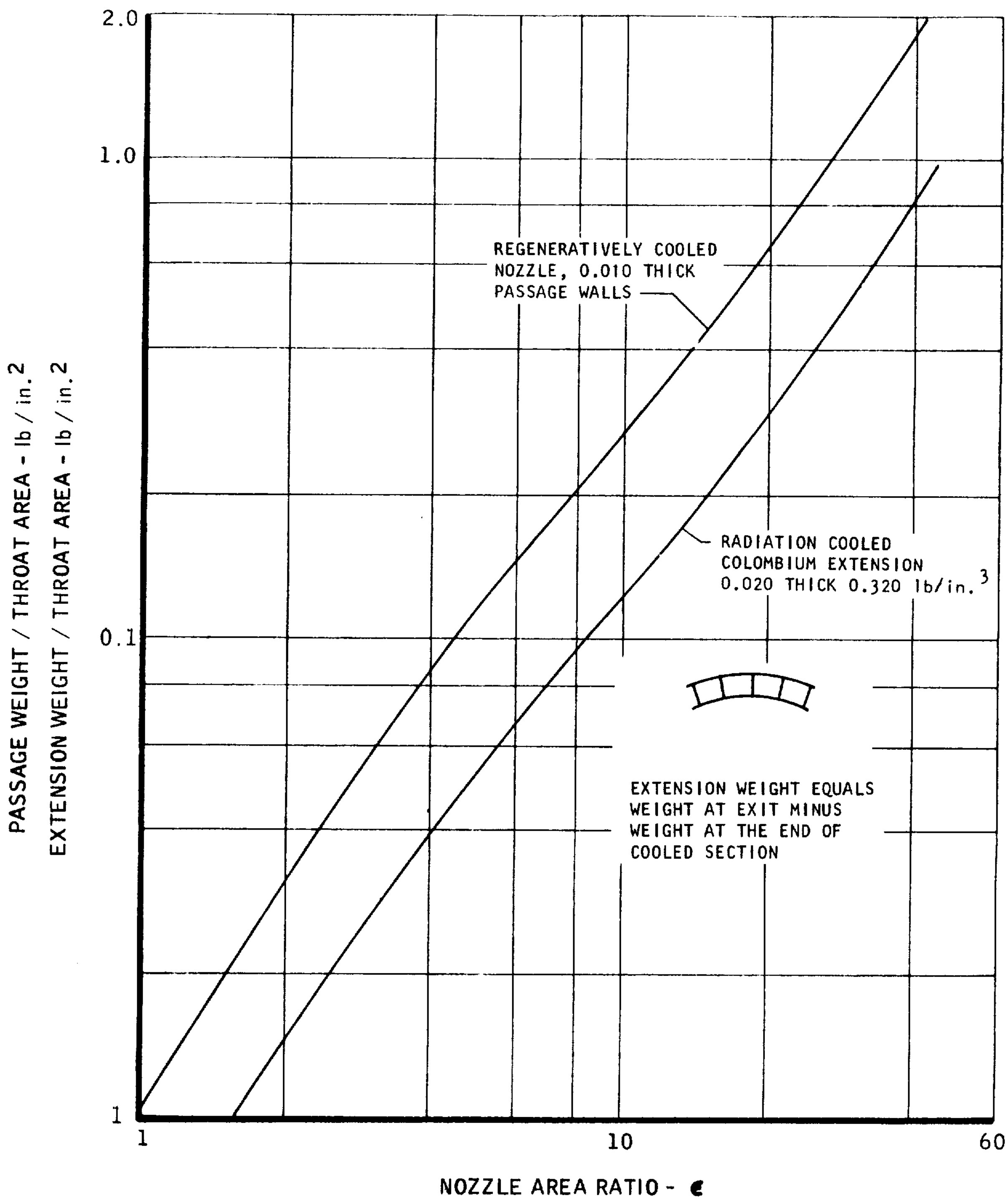
REINFORCEMENT WEIGHT DOWNSTREAM FROM NOZZLE THROAT



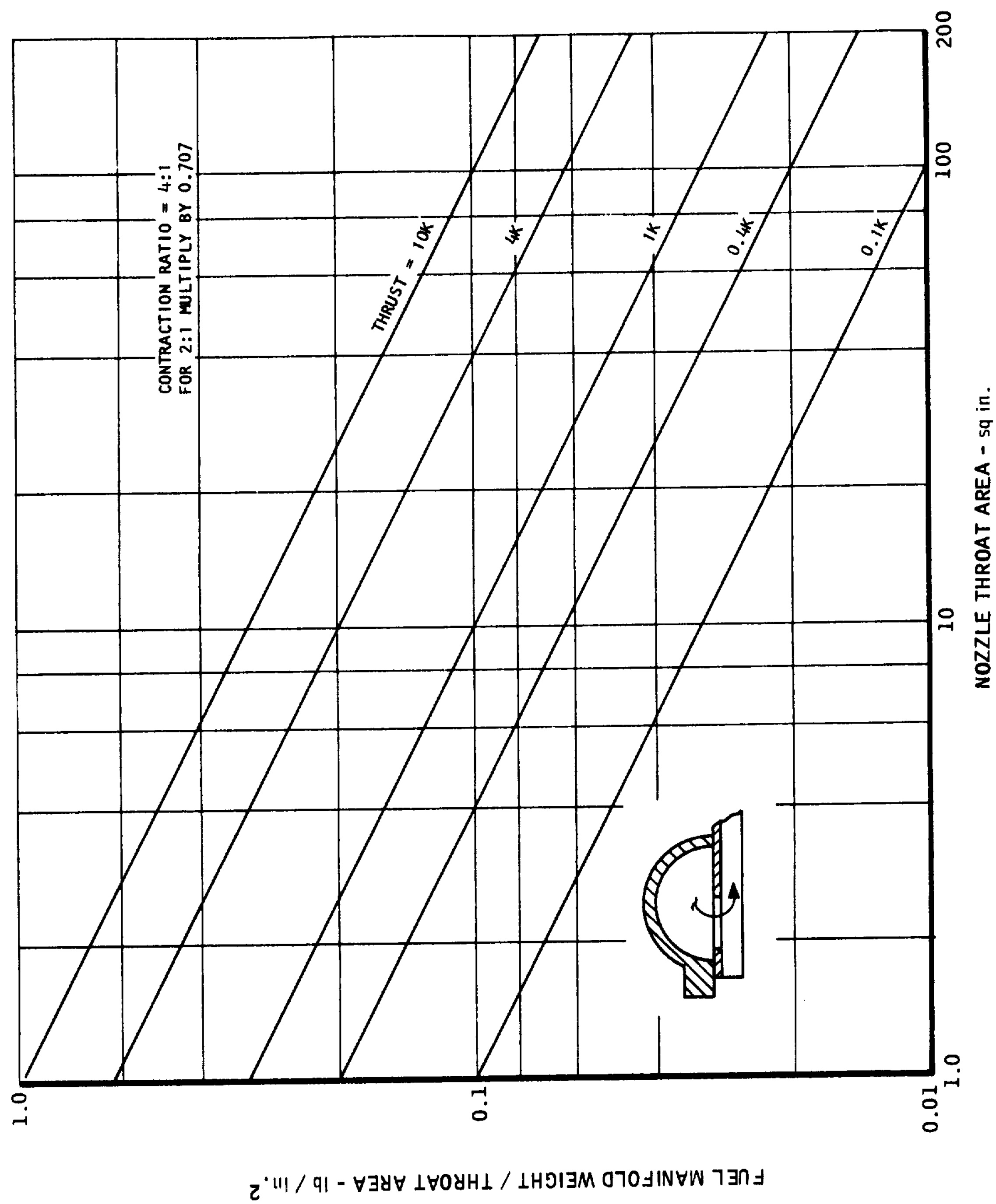
COOLANT PASSAGE WEIGHT UPSTREAM FROM NOZZLE THROAT



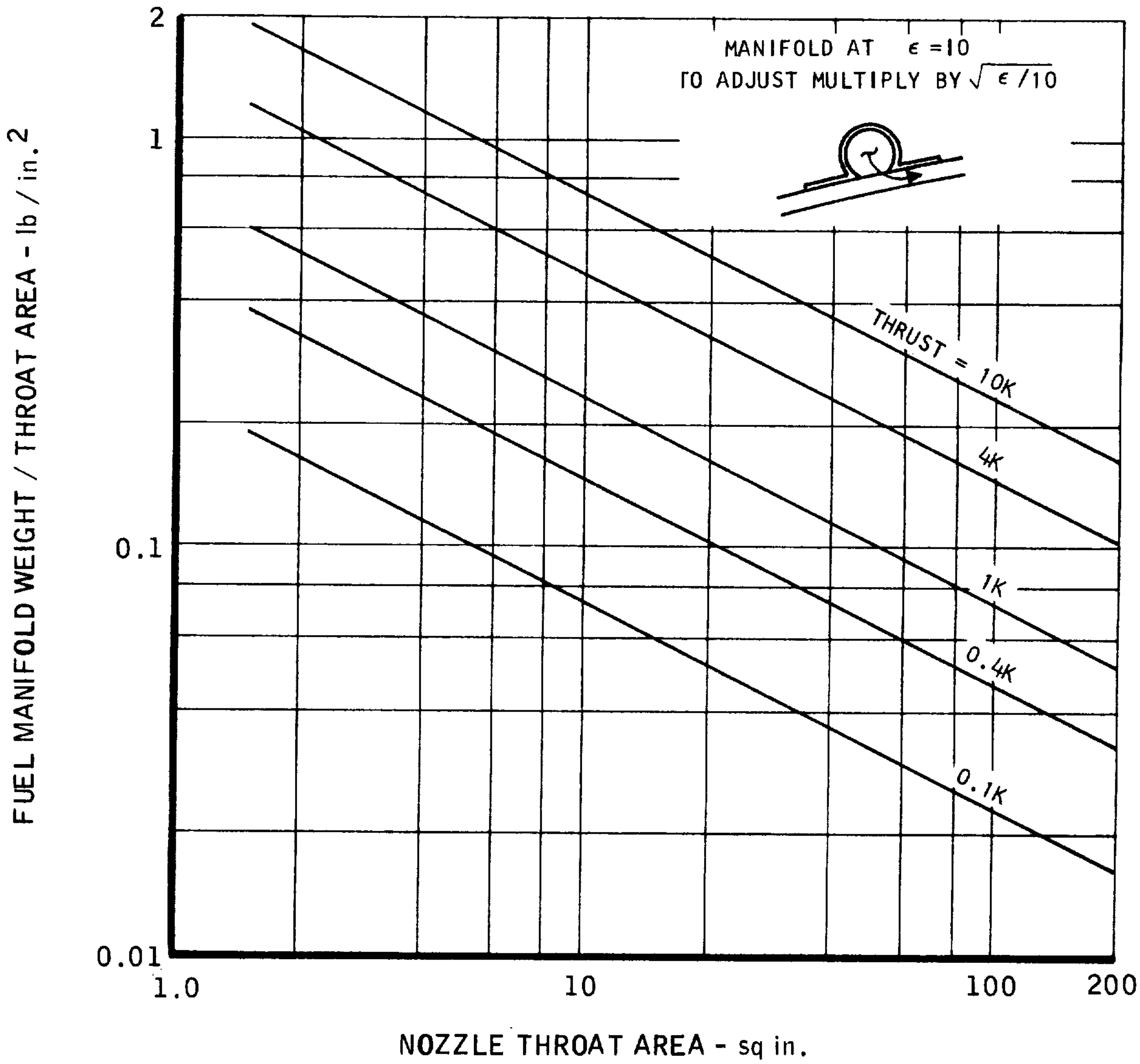
REGENERATIVE COOLANT PASSAGE AND EXTENSION WEIGHT
DOWNSTREAM FROM NOZZLE THROAT TO NOZZLE EXIT PLANE

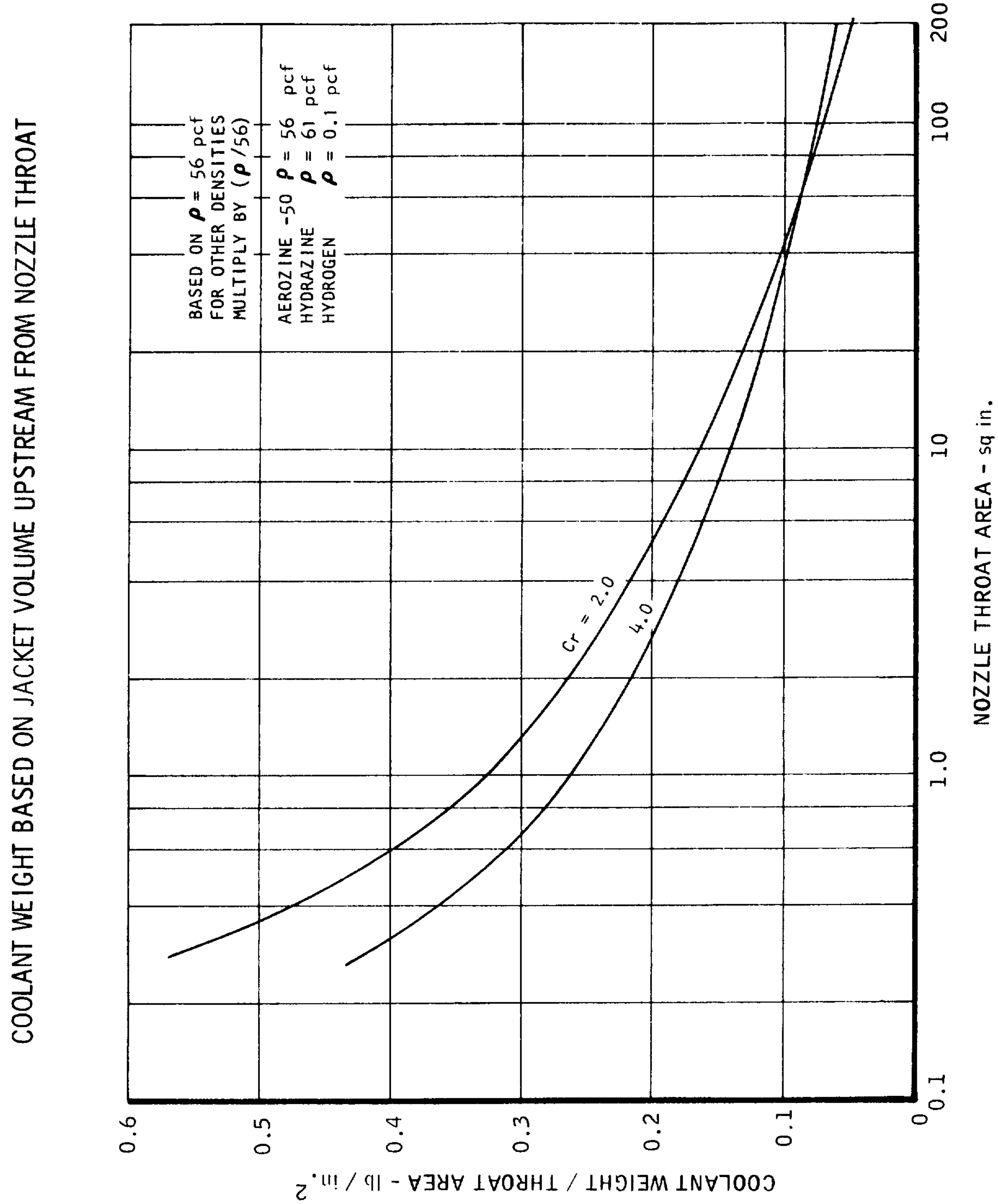


FUEL MANIFOLD WEIGHT FOR N_2H_4 AND AEROZINE-50 COOLED CHAMBERS

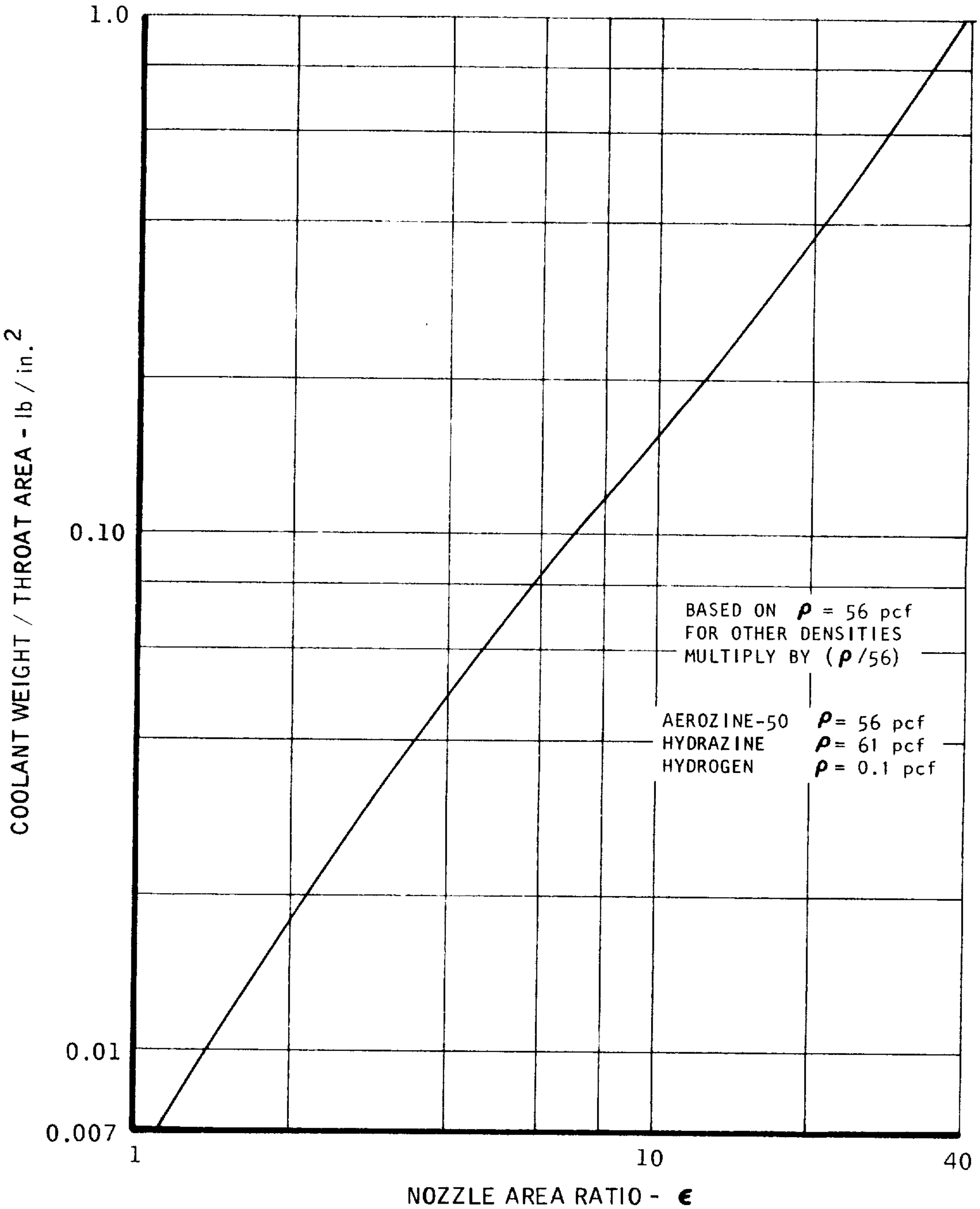


FUEL MANIFOLD WEIGHT FOR HYDROGEN COOLED CHAMBERS

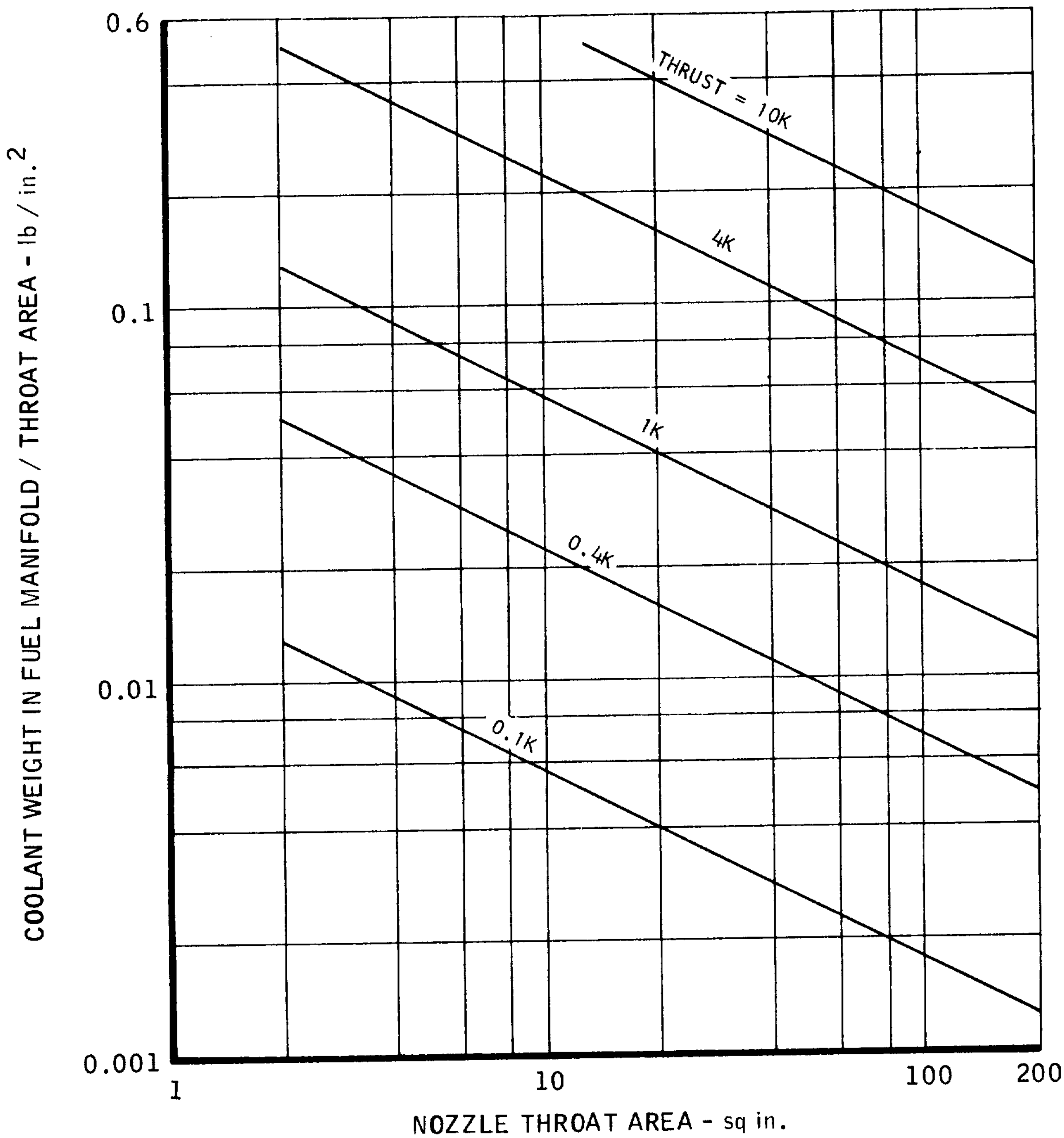



COOLANT WEIGHT BASED ON JACKET VOLUME UPSTREAM FROM NOZZLE THROAT
BASED ON ρ = 56 pcf
FOR OTHER DENSITIES
MULTIPLY BY (ρ/56)
AEROZINE -50 ρ = 56 pcf
HYDRAZINE ρ = 61 pcf
HYDROGEN ρ = 0.1 pcf
Cr = 2.0
4.0
0
0.1
0.2
0.3
0.4
0.5
0.6
COOLANT WEIGHT / THROAT AREA - lb / in.2
0.1
1.0
10
100
200
NOZZLE THROAT AREA - sq in.

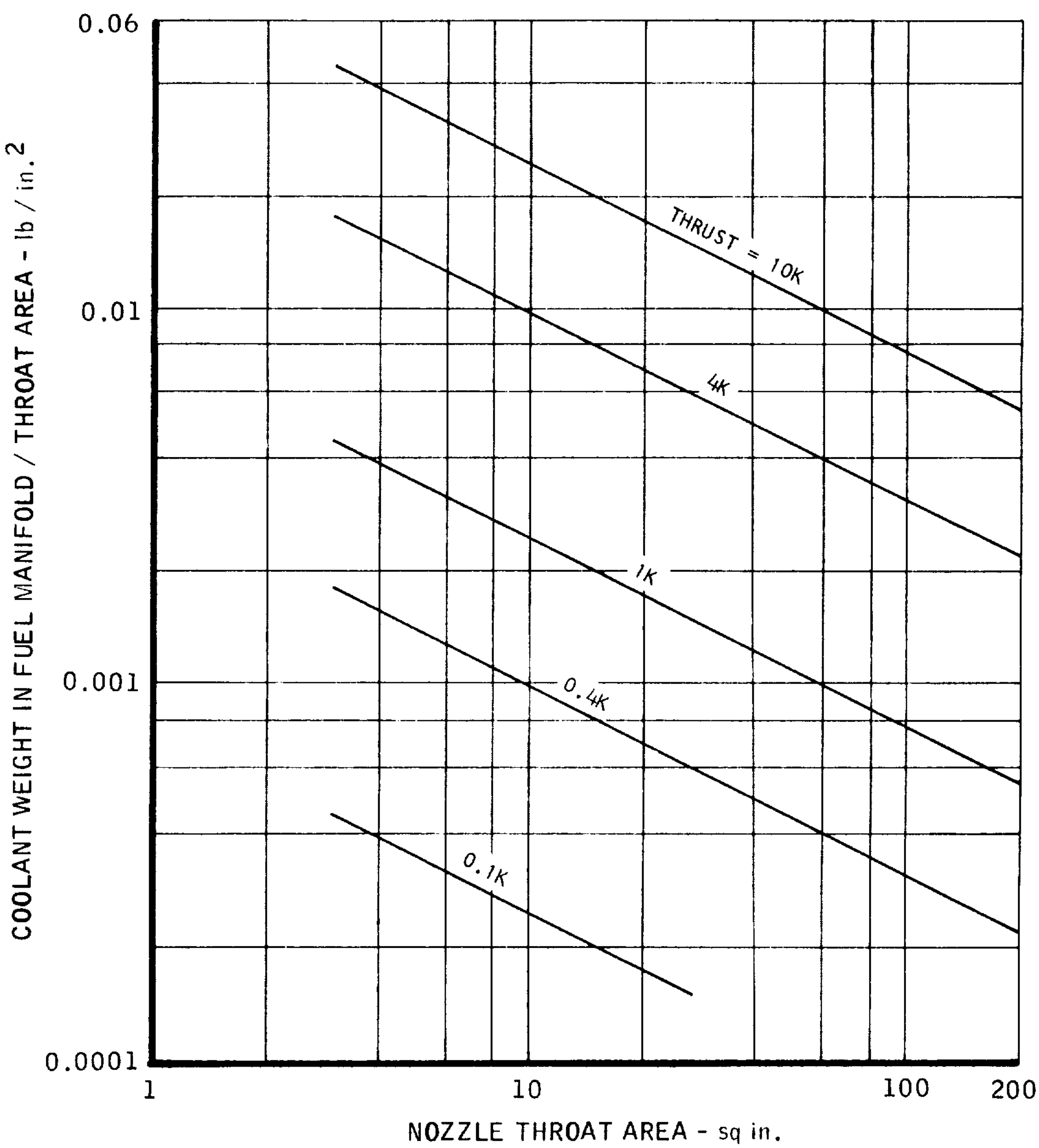
COOLANT WEIGHT BASED ON JACKET VOLUME DOWNSTREAM FROM NOZZLE THROAT

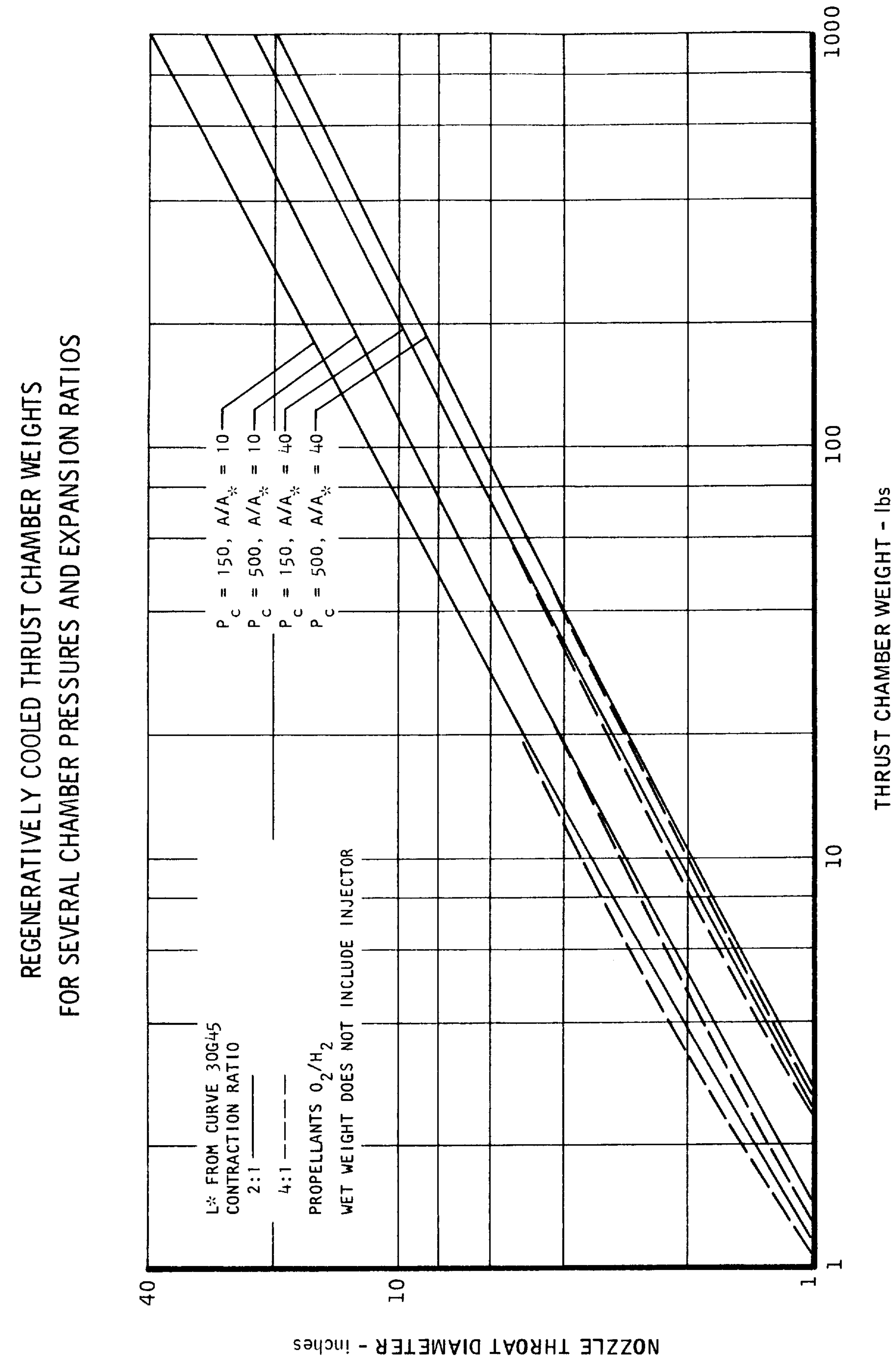


COOLANT WEIGHT BASED ON FUEL MANIFOLD VOLUME FOR AEROZINE-50 AND N_2H_4

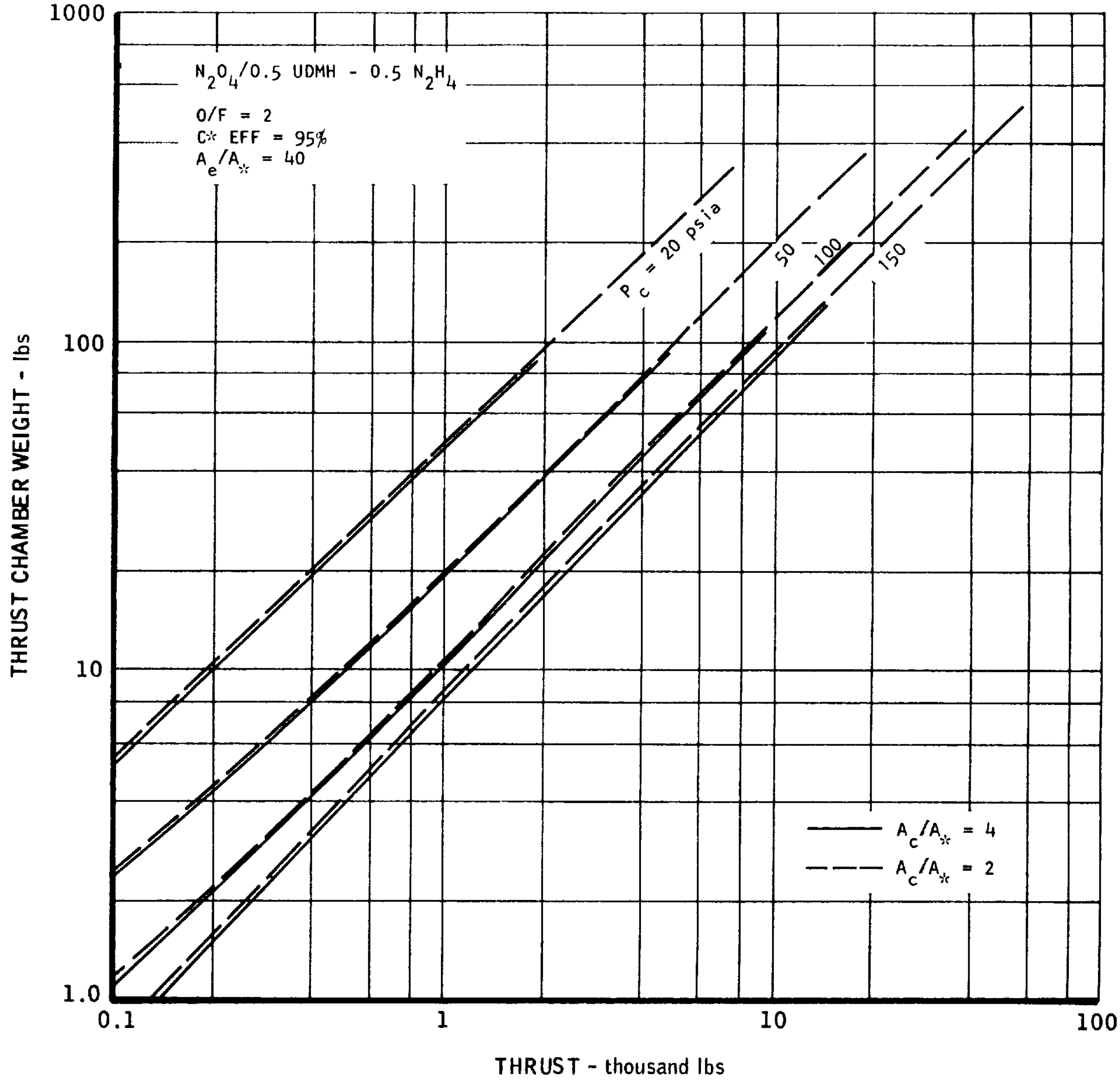


COOLANT WEIGHT BASED ON FUEL MANIFOLD VOLUME FOR HYDROGEN

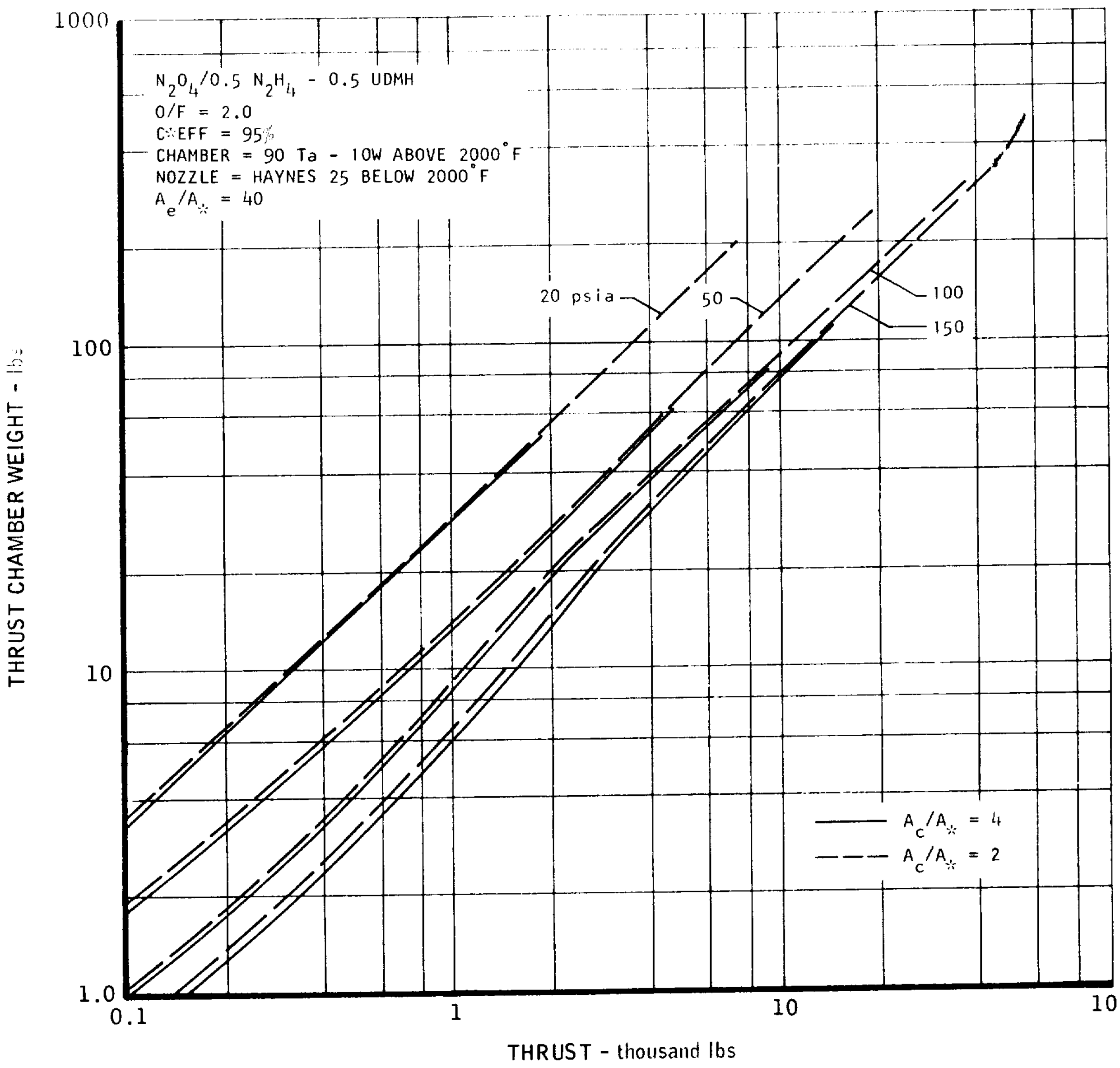



REGENERATIVELY COOLED THRUST CHAMBER WEIGHTS
FOR SEVERAL CHAMBER PRESSURES AND EXPANSION RATIOS
P_c = 150, A/A_* = 10
P_c = 500, A/A_* = 10
P_c = 150, A/A_* = 40
P_c = 500, A/A_* = 40
L* FROM CURVE 30G45
CONTRACTION RATIO
2:1
4:1
PROPELLANTS O_2/H_2
WET WEIGHT DOES NOT INCLUDE INJECTOR
THRUST CHAMBER WEIGHT - lbs
1
10
100
1000
NOZZLE THROAT DIAMETER - inches
40
10
1

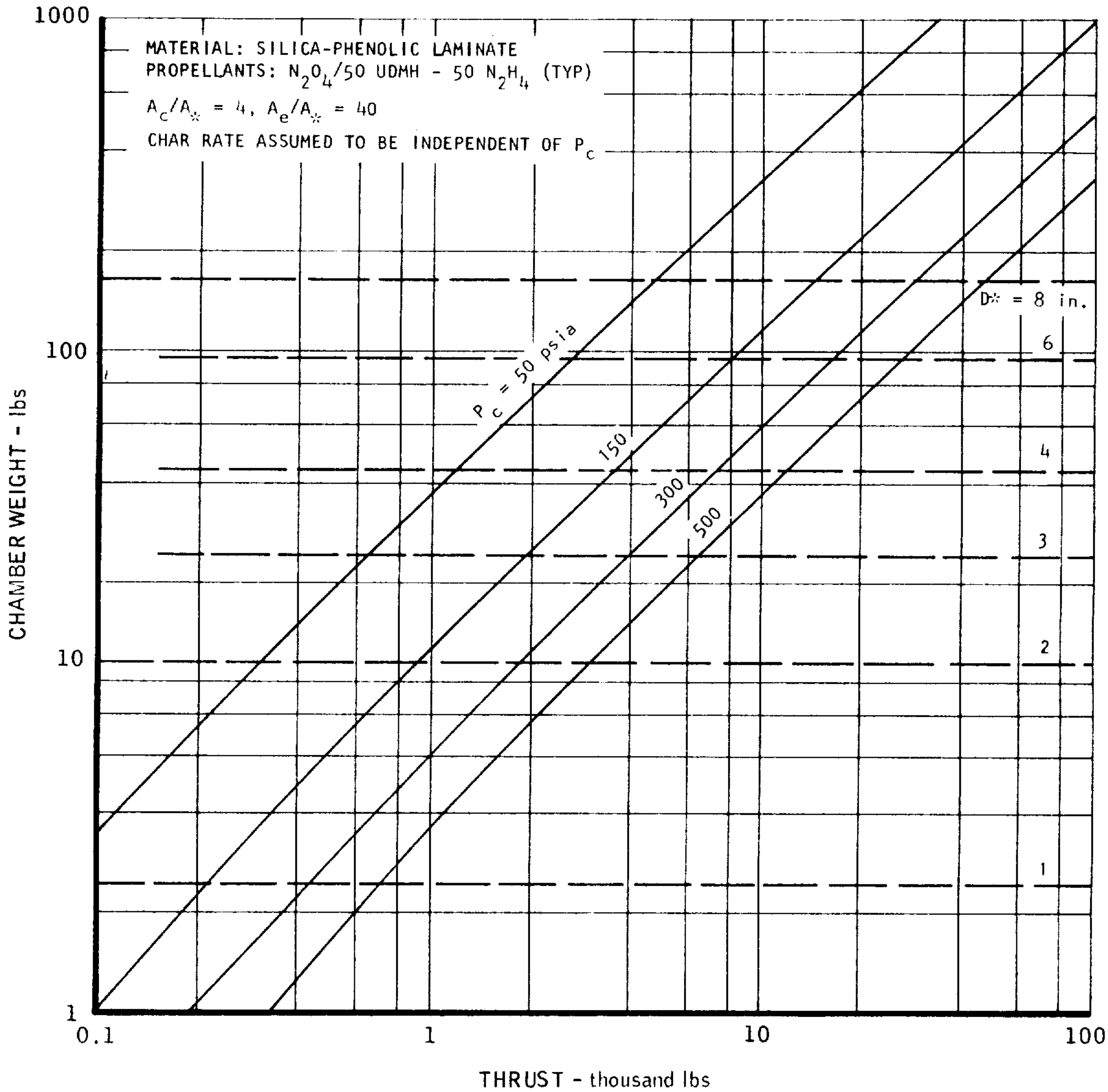
WEIGHT OF RADIATION COOLED THRUST CHAMBER USING 90 Ta-10 W ALLOY



WEIGHT OF RADIATION COOLED THRUST CHAMBER USING 90 Ta-10W AND HAYNES 25 ALLOYS



CHARACTERISTIC ABLATIVE THRUST CHAMBER WEIGHT
AS A FUNCTION OF THRUST FOR 60 - second STEADY STATE RUN



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CHARACTERISTIC ABLATIVE THRUST CHAMBER WEIGHT
AS A FUNCTION OF THRUST FOR 300 - second STEADY STATE RUN

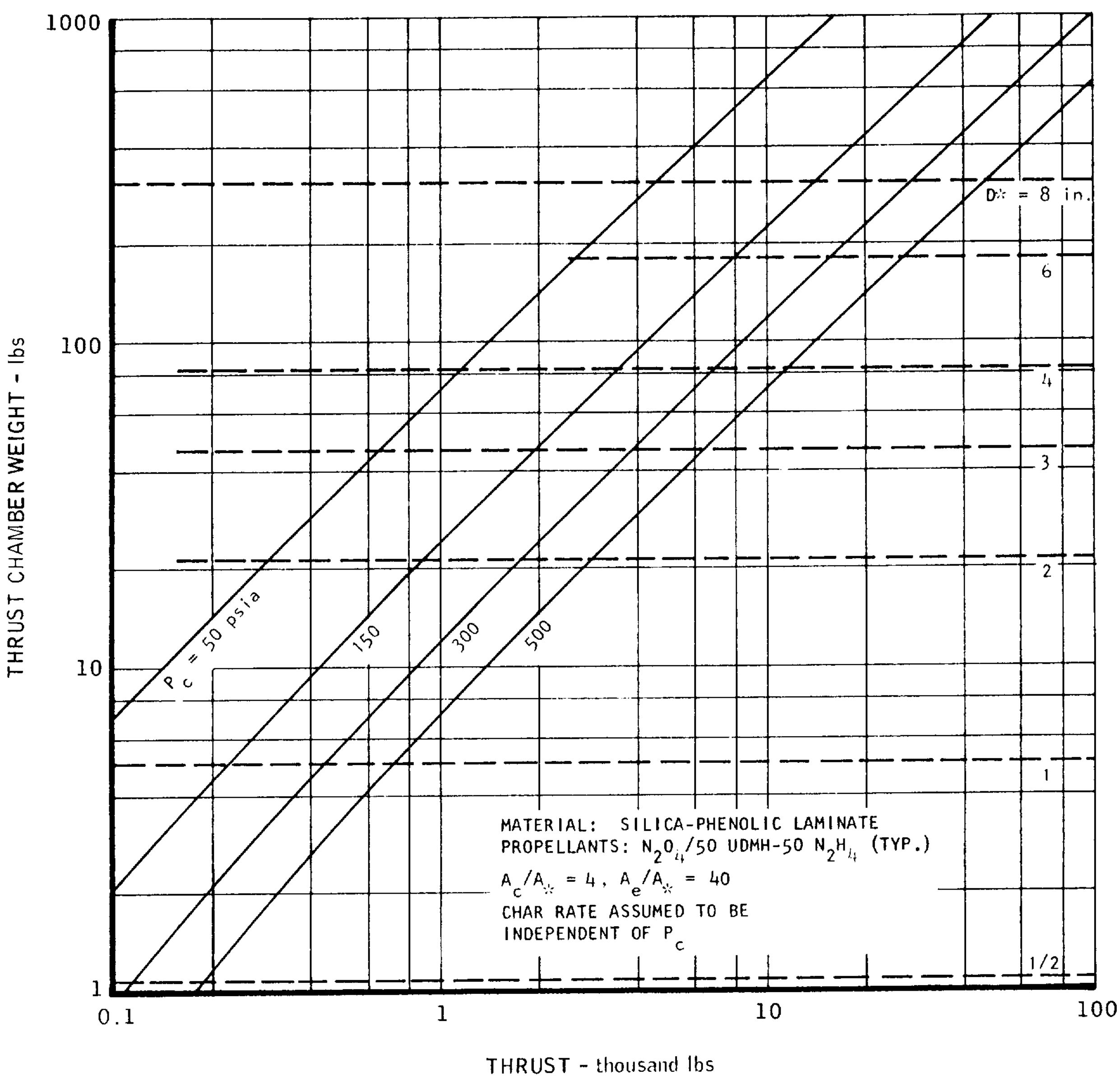
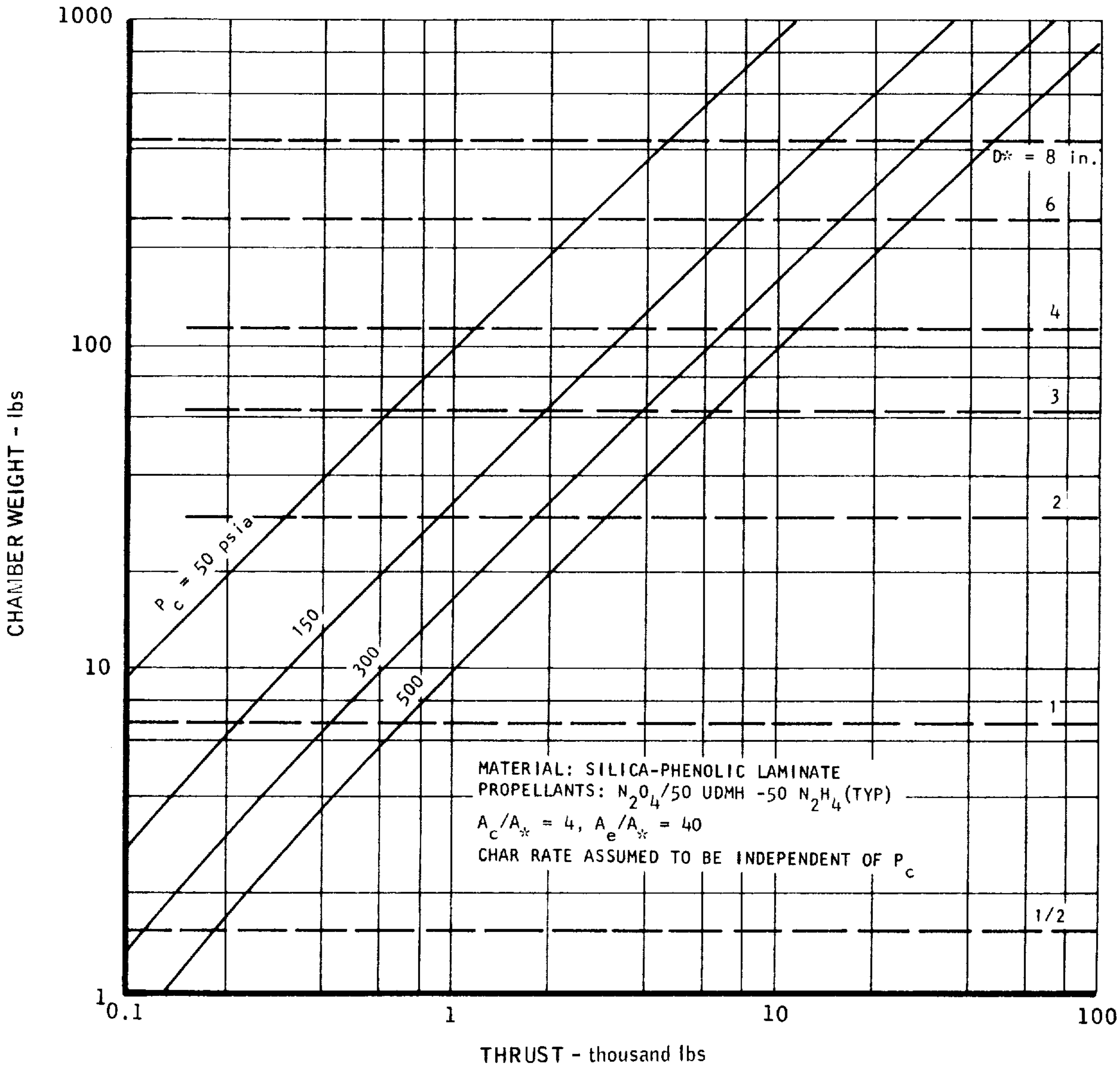


FIGURE 38

CHARACTERISTIC ABLATIVE THRUST CHAMBER WEIGHT
AS A FUNCTION OF THRUST FOR 600 - second STEADY STATE RUN



CHARACTERISTIC ABLATIVE THRUST CHAMBER WEIGHT
AS A FUNCTION OF THRUST FOR 1000 - second STEADY STATE RUN

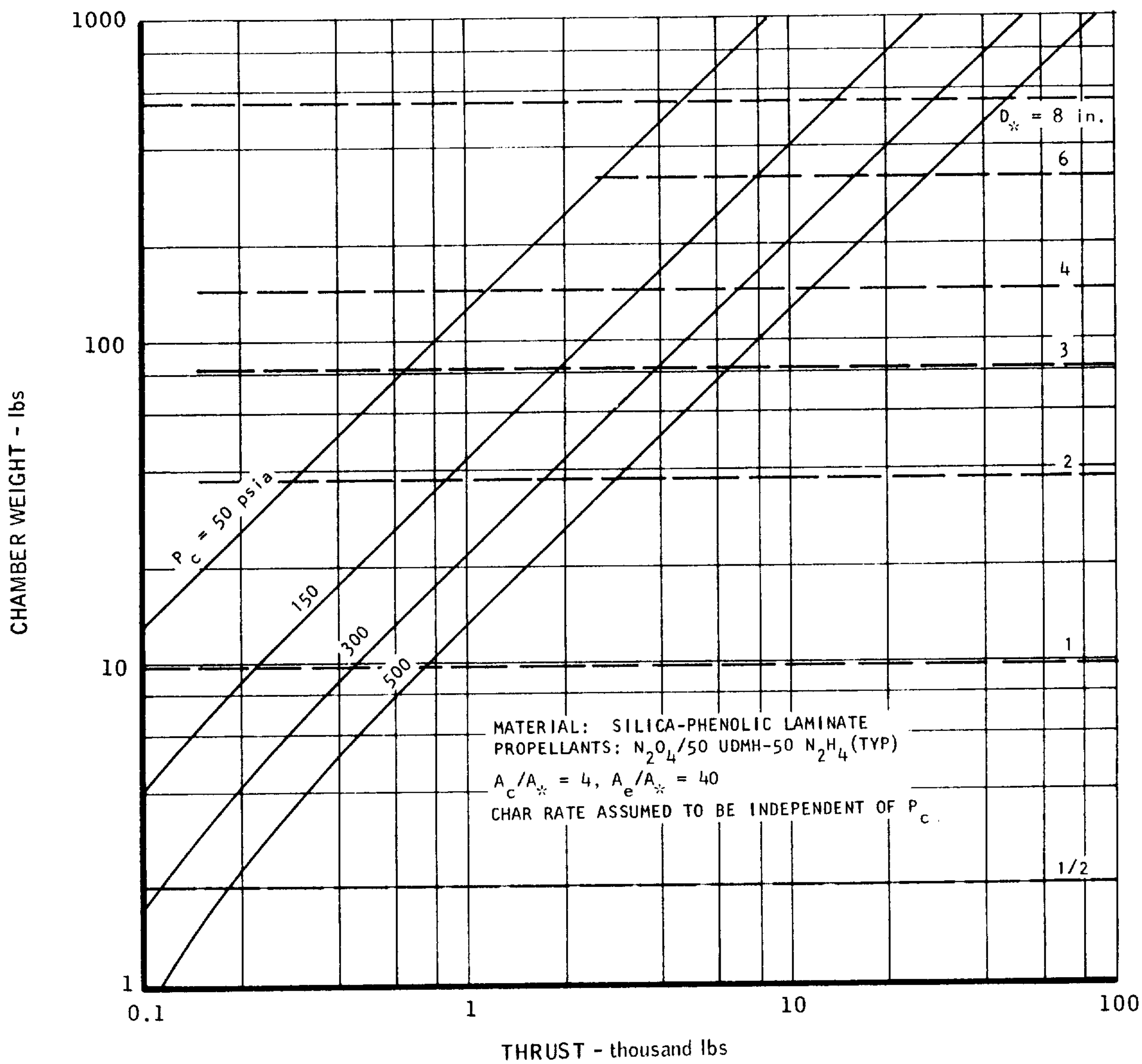
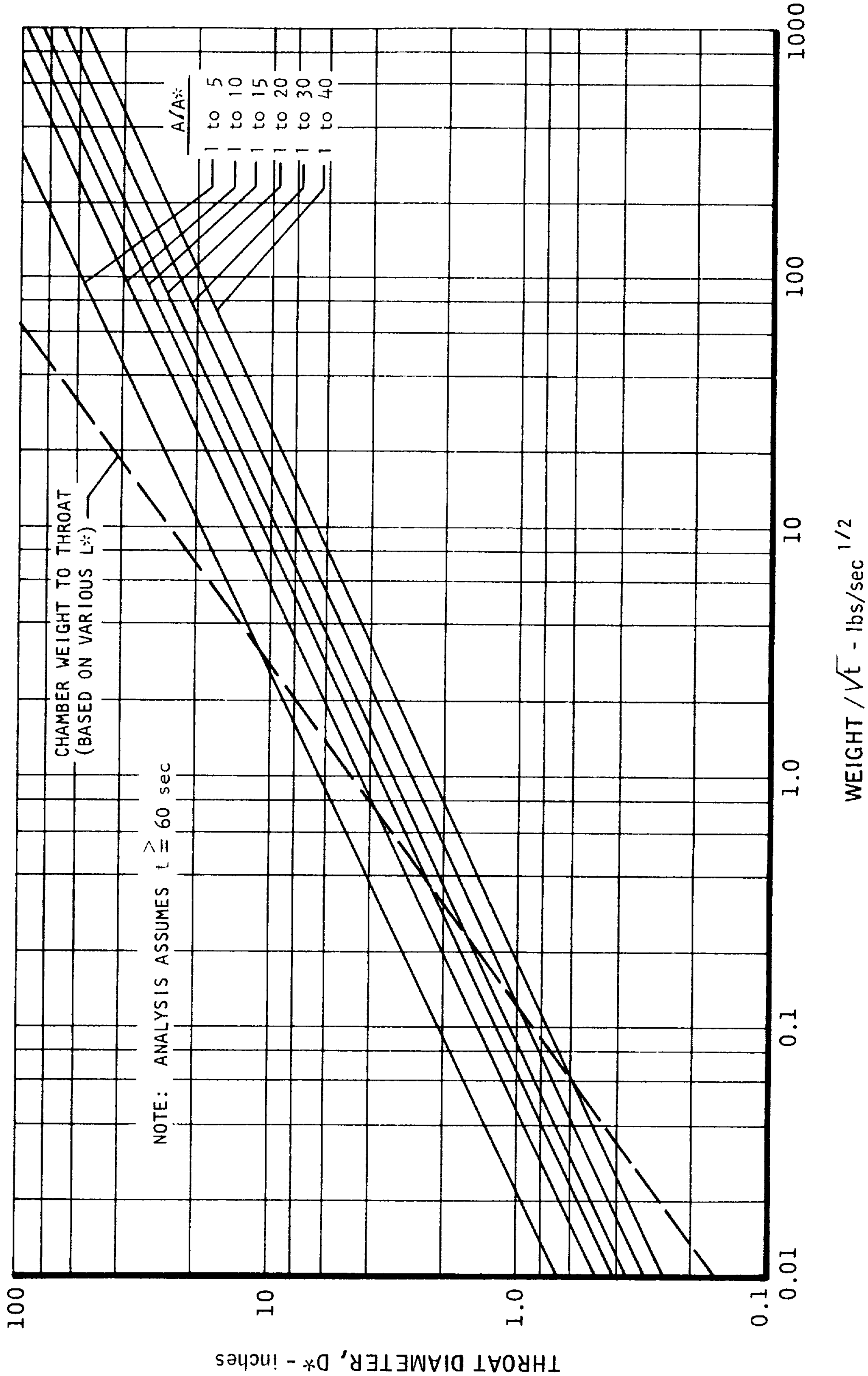
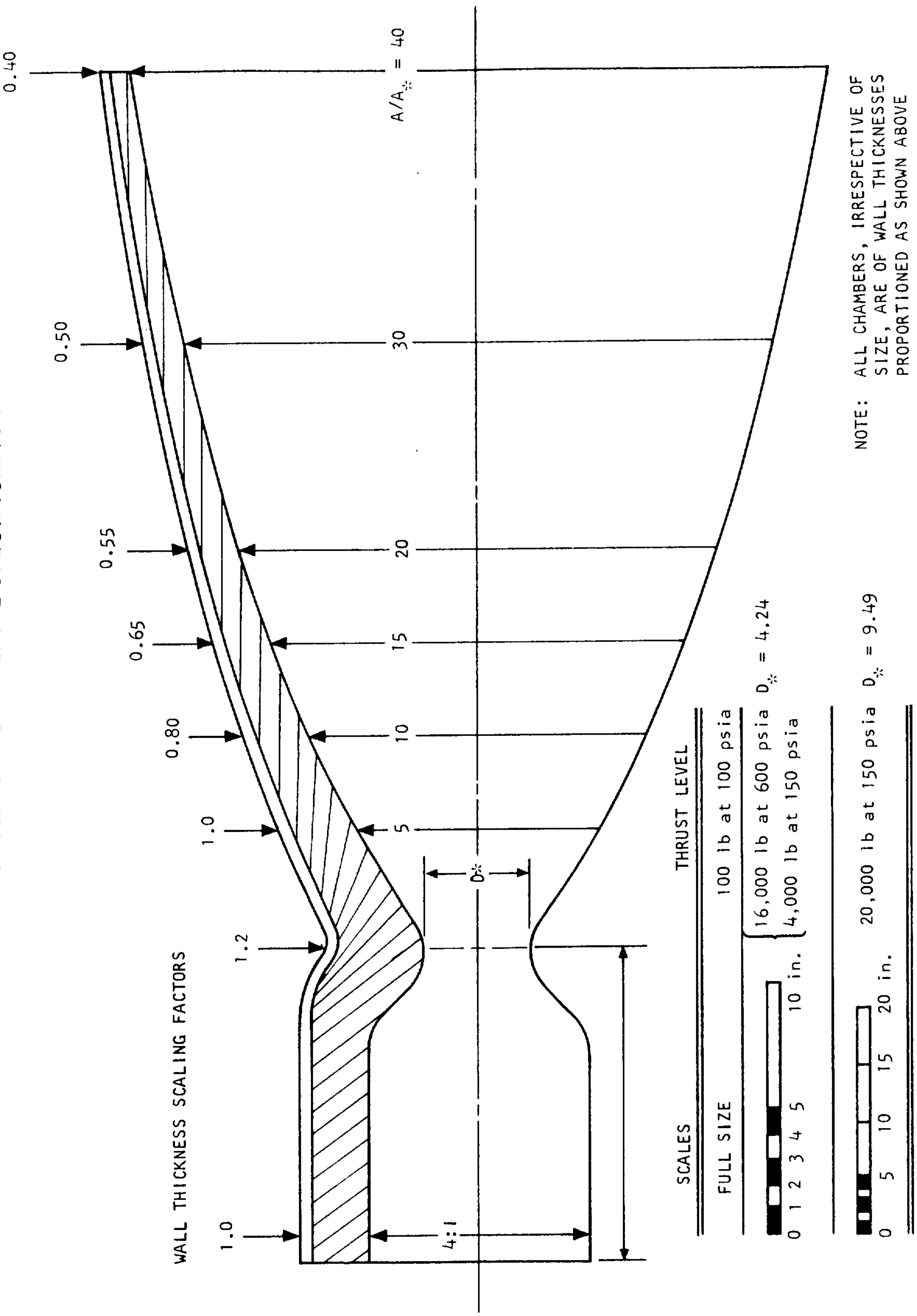


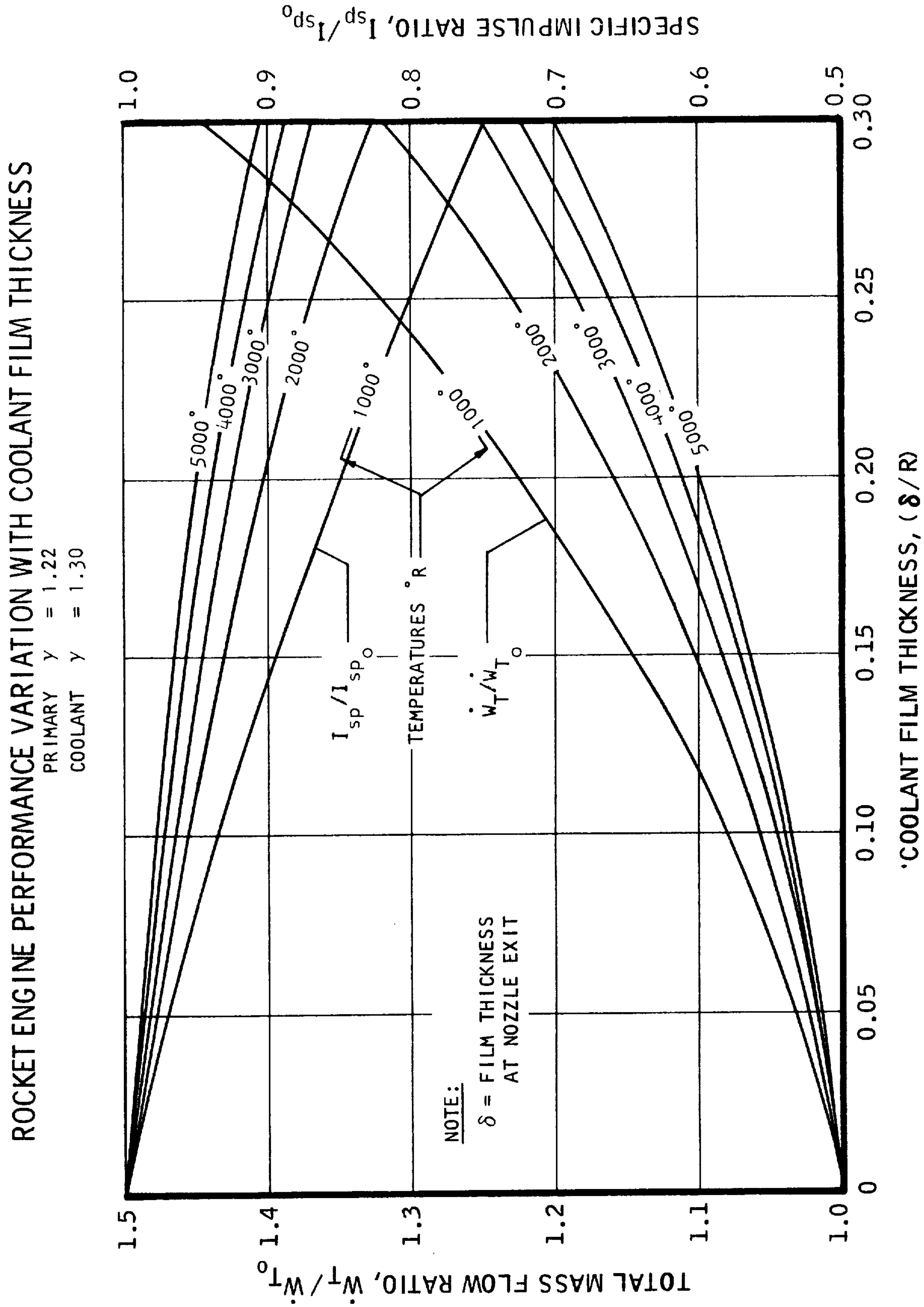
FIGURE 40

VARIATION OF ABLATIVE THRUST CHAMBER WEIGHT PARAMETER WITH THROAT DIAMETER

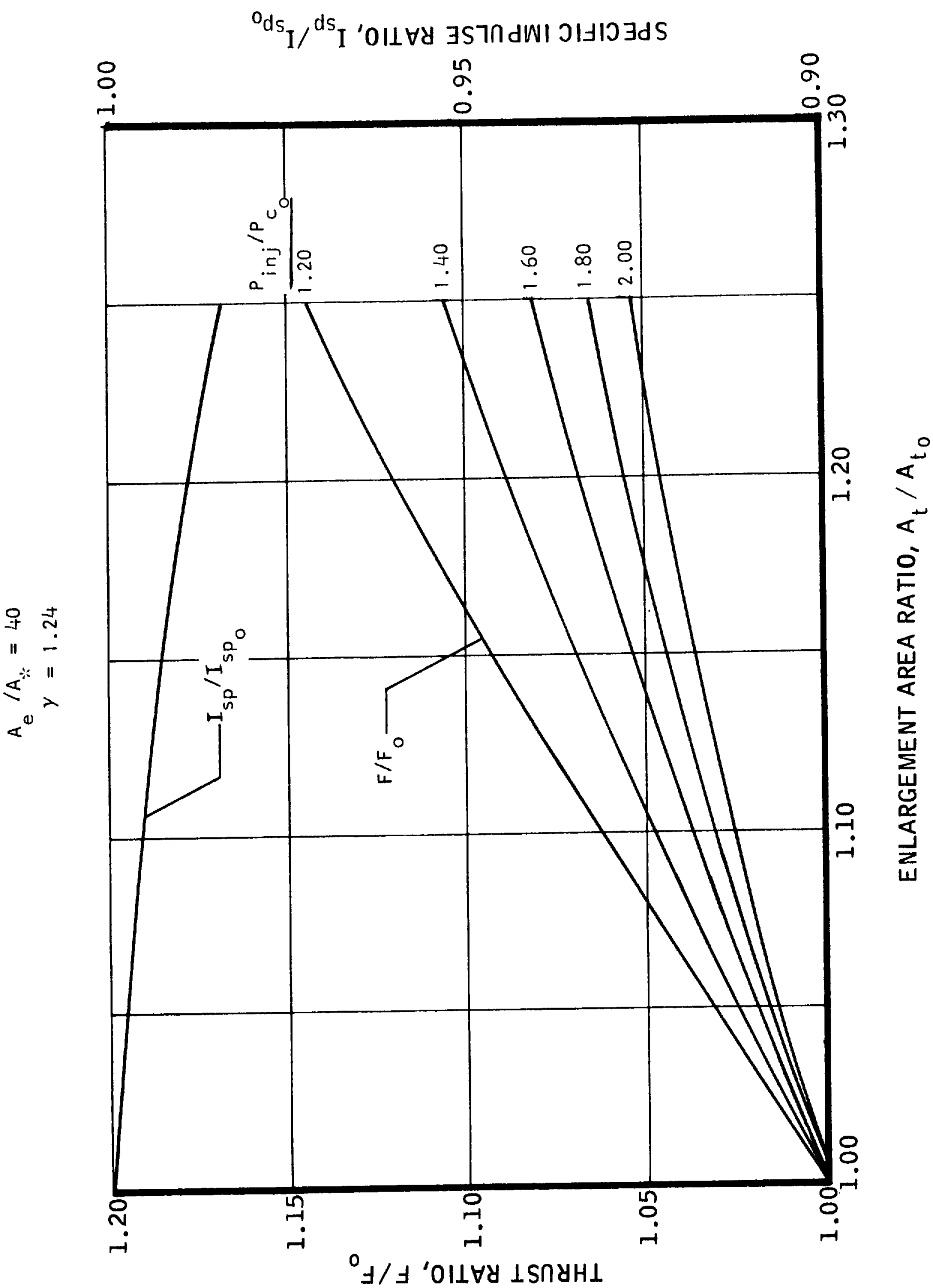


DESIGN LAYOUT FOR WEIGHT ANALYSIS OF A TYPICAL ABLATIVE THRUST CHAMBER DESIGN
FOR COOLING AND MATERIAL SPECIFICATION

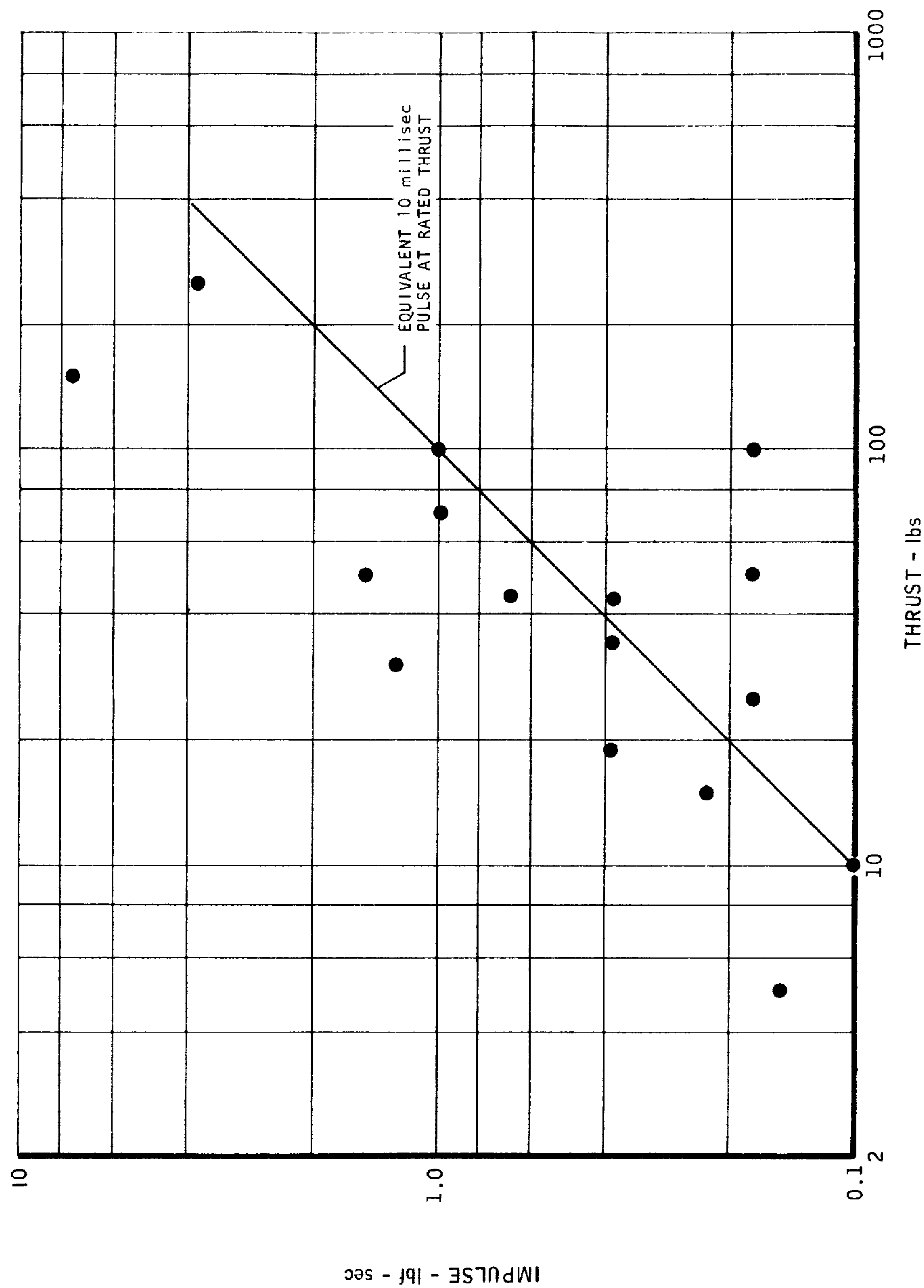



ROCKET ENGINE PERFORMANCE VARIATION WITH COOLANT FILM THICKNESS
PRIMARY γ = 1.22
COOLANT γ = 1.30
SPECIFIC IMPULSE RATIO, I_{sp}/I_{sp_o}
1.0
0.9
0.8
0.7
0.6
0.5
TOTAL MASS FLOW RATIO, $\dot{W}_T/\dot{W}_{T_o}$
1.5
1.4
1.3
1.2
1.1
1.0
COOLANT FILM THICKNESS, (δ/R)
0
0.05
0.10
0.15
0.20
0.25
0.30
5000°
4000°
3000°
2000°
1000°
1000°
2000°
3000°
4000°
5000°
I_{sp}/I_{sp_o}
TEMPERATURES °R
$\dot{W}_T/\dot{W}_{T_o}$
NOTE:
δ = FILM THICKNESS AT NOZZLE EXIT

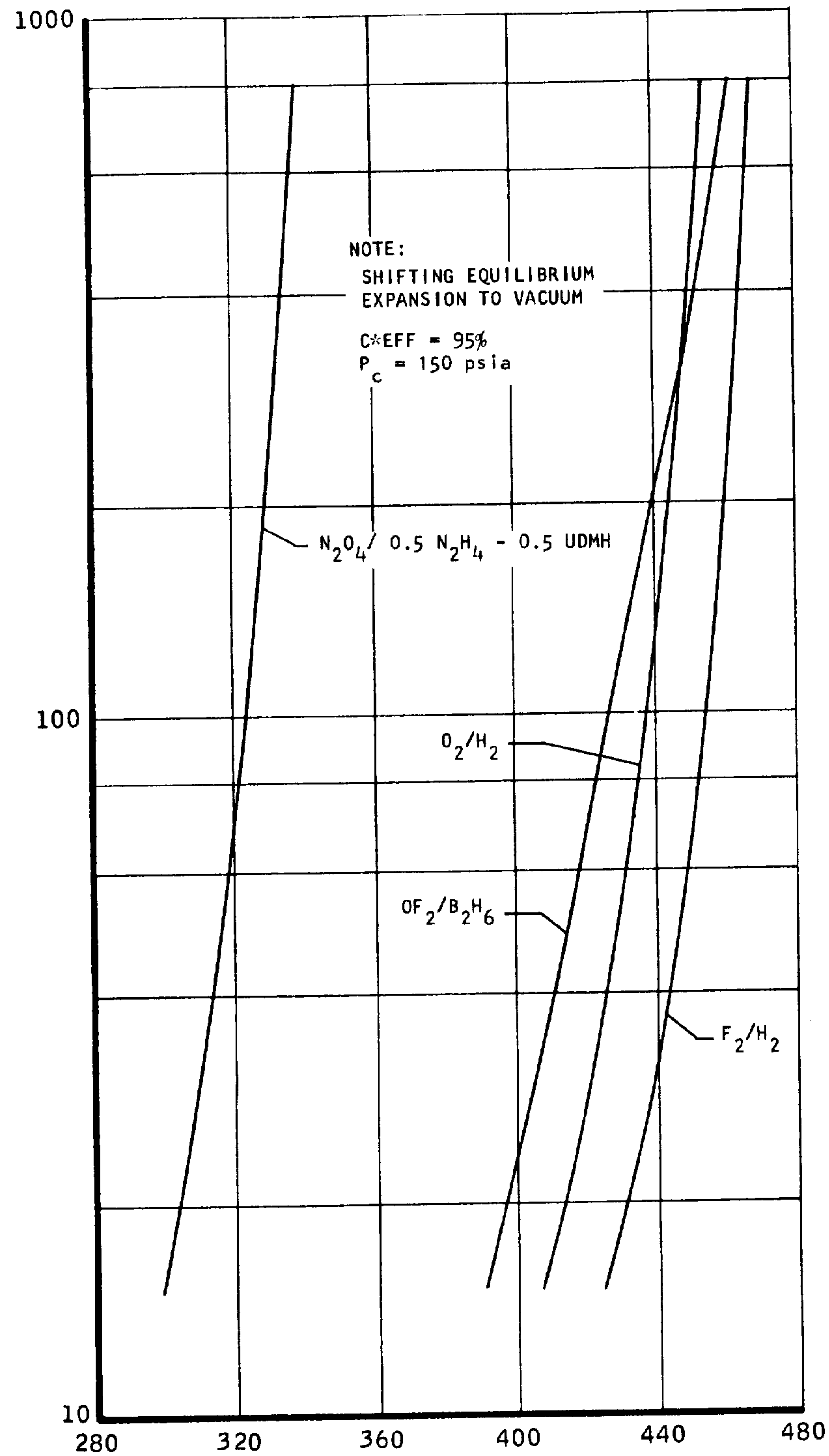
VARIATION OF PERFORMANCE PARAMETER DUE TO ROCKET NOZZLE THROAT ENLARGEMENT



VARIATION OF TYPICAL REQUIRED MINIMUM IMPULSE BITS WITH THRUST LEVEL
FOR ATTITUDE CONTROL ROCKET ENGINES

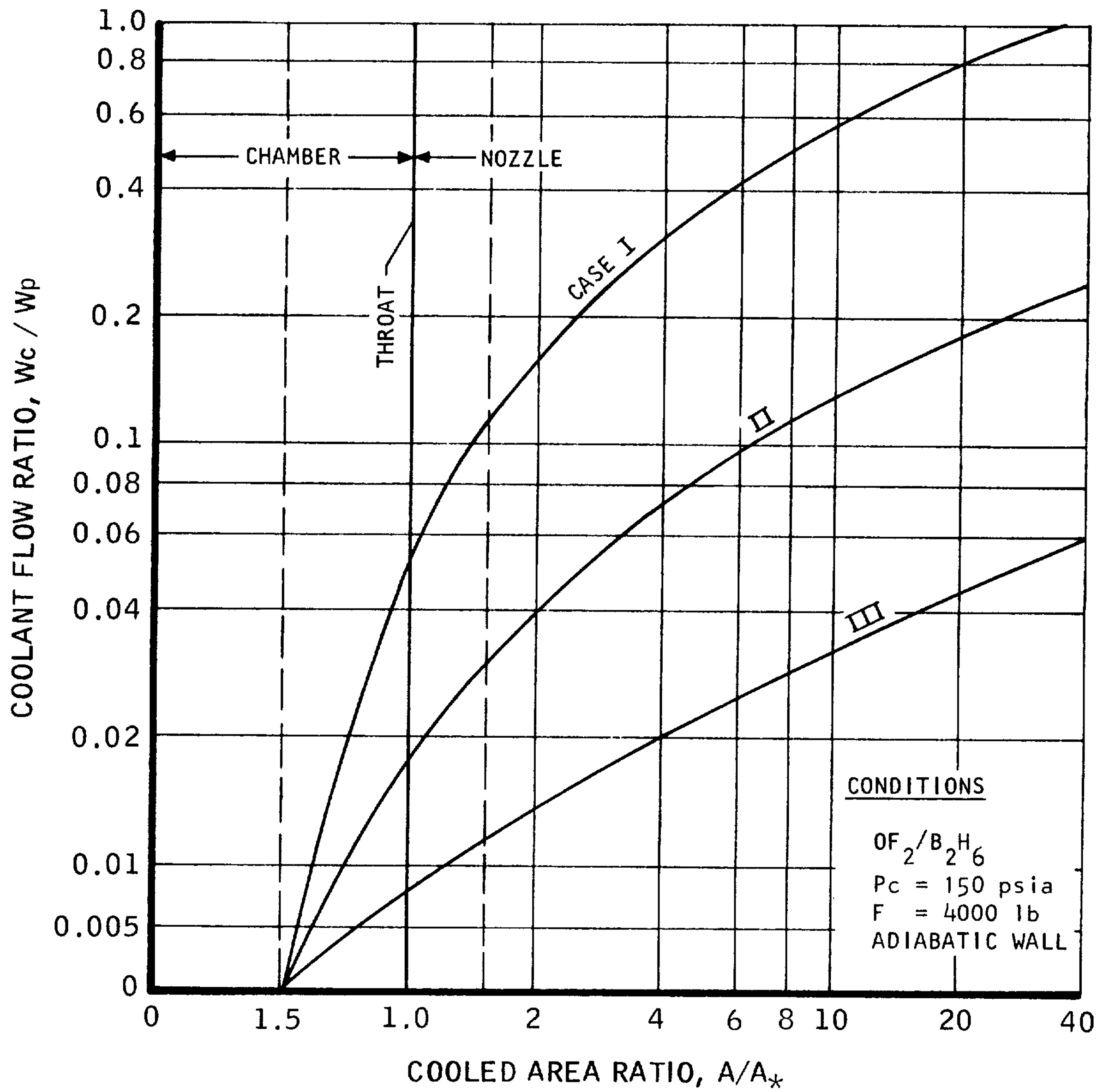


VARIATION OF SPECIFIC IMPULSE WITH NOZZLE EXPANSION RATIO



FILM COOLING REQUIREMENT FOR TOTAL SURFACE AREA
FROM AHEAD OF THROAT (Ac/A* 1.5) TO AREA RATIO INDICATED

CASE I: LIQUID FILM CONTINUOUS
II: GAS FILM (JPC RPT. I-62-2) Tw = 2200°R
III: VAPOR FILM (JPC RPT. TM62-5) Tw = 2200°R



APPENDIX A

SUMMARY OF NOMENCLATURE

Symbol	Description	Units
A_*	Rocket nozzle throat area	in.^{2}
A_c	Combustion chamber cross section area	in.^{2}
A_e	Nozzle area at exit plane	in.^{2}
A_t	Rocket nozzle throat area	in.^{2}
C*	Characteristic velocity	ft/sec
C_F	Rocket nozzle thrust coefficient	--
c_p	Specific heat at constant pressure	Btu/lb °F
C_r	Contraction ratio A_e/A_*	--
C-D	Refers to convergent-divergent rocket nozzle contour	--
D*	Nozzle throat diameter	in.
EDA	Ethylenediamine	--
F	Thrust (pounds force)	lbf
g	Gravitational constant	ft/sec^2
h	Heat transfer coefficient	Btu/hr ft^2 °F
I_{sp}	Specific Impulse $F/\dot{W}_p$	lbf-sec/lbm
I_t	Total impulse	--
k	Thermal conductivity	Btu/hr ft °F
L*	Characteristic combustion chamber length $L^* = \frac{V_c}{A_*}$	in.
L_n	Length of rocket nozzle from throat to exit plane	in.
MMH	Monomethylhydrazine	--
M_f	Final mass	lbm

APPENDIX A (Continued)

Symbol	Description	Units
M_o	Initial mass	lbm
M_{PL}	Payload mass	lbm
O/F	Oxidizer to fuel mass flow ratio	--
P_a	Ambient pressure	psia
P_c	Combustion chamber pressure	psi
P_{sup}	Propellant supply pressure	psia
q/A	Heat flux	Btu/in.^{2} sec
t	Time	sec
T_g	Gas temperature	°F
T_w	Wall temperature	°F
90 Ta-10W	Refractory metal alloy of 90% tantalum-10% tungsten	--
UDMH	Unsymmetrical Dimethylhydrazine	--
ΔV	Velocity increment	ft/sec
V_c	Combustion chamber volume	in.^{3}
W_c	Coolant weight	lbm
W_p	Propellant weight	lbm
$\dot{W}_p$	Propellant flow rate (pounds mass per second)	lbm/sec
W_s	Structure weight	lbm
$\frac{W_{initial}}{W_{final}}$	Ratio of initial to final weight of spacecraft resulting from propellant expenditure	--

APPENDIX A (Continued)

Symbol	Description	Units
Zero g	Zero effective gravitational force	--
δ	Film thickness	in.
ε	Nozzle expansion ratio A_e/A_*	--
γ	Specific heat ratio	--
ρ	Density	lb/ft^3

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